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ISSUE NO.	ISSUED TO

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Abstract and Key-Words

Abstract

A description of the Lunar Orbiter spacecraft and each of the subsystems is outlined in this document.

A brief description of a typical mission, the resulting data and the on-line supporting ground based equipment is given.

Key Words

Lunar Orbiter

Photography

Spacecraft

Camera

Photographic Data

Selenography

Selenodesy

Video

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1.0 PURPOSE AND SCOPE

This document is the second volume of the final report required by contract NAS1-4959, "A Research Study of the Lunar Orbiter Spacecraft Regarding its Adaptability to Other Scientific Investigations". It is intended to provide a background for the study, giving a general, yet comprehensive, description of the Block I spacecraft and its subsystems.

2.0 REFERENCE DOCUMENTS

None

3.0 DESCRIPTION

3.1 Introduction

3.1.1 Mission Objectives

The primary objective of the Lunar Orbiter is to obtain topographic data about the lunar surface. This information is necessary for the selection and confirmation of landing sites for Apollo. It will also extend the scientific knowledge of the moon's surface. Secondary mission objectives are to secure information about the gravitational field and certain lunar environmental data. Specific objectives for the mission are as follows:

a. Take moderate-resolution photographs of at least 40,000 square kilometers of lunar surface about the prime target located at 2.5° south latitude, 36° west longitude.

b. Take high-resolution photographs of at least 8,000 square kilometers of lunar surface at the target located at 2.3° south latitude, 53° west longitude.

c. Obtain lunar environmental data on micrometeoroid flux and energetic particle flux.

d. Transmit all photographic and environmental data to earth.

e. Establish lunar orbits which will provide the basis for ascertaining the lunar gravitational potential and obtaining selenodetic information from earth-based tracking data.

3.1.2 Mission Description

3.1.2.1 Launch Vehicle

A two-stage launch vehicle, consisting of an Atlas D (model SLV-3) first stage and an Agena D (Model SS01B) second stage will boost the spacecraft into a translunar trajectory via an earth-parking orbit. The launch vehicle configuration is shown in Figure 3.1.2-1.

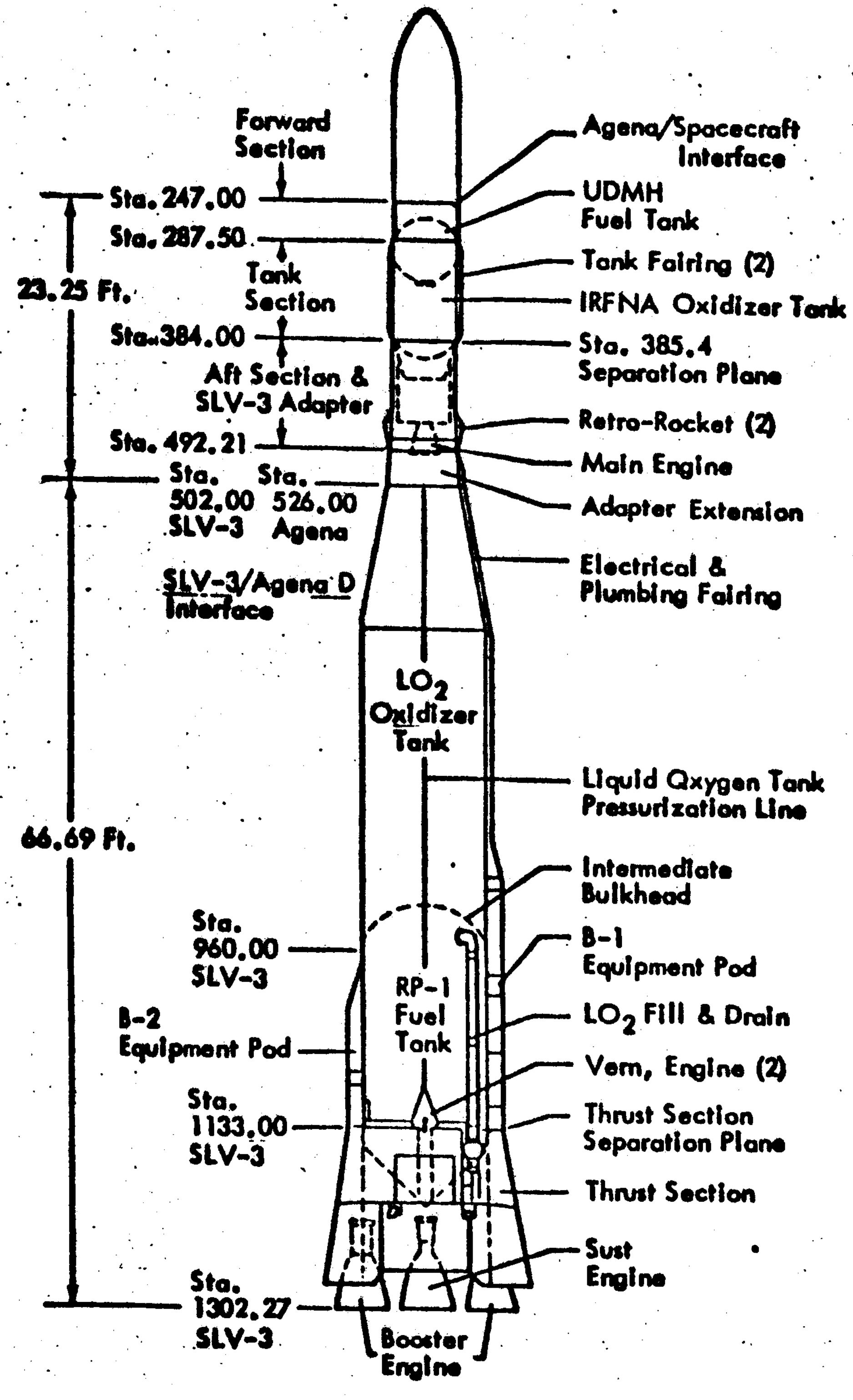


FIGURE 3.1.2-1

LAUNCH VEHICLE CONFIGURATION

3.1.2.1.1 SLV-3 Configuration

The SLV-3 airframe consists of two major sections: booster and sustainer. Engines are gimbal-mounted and swiveled for directional control and stability. They are ignited prior to launch and are all fed from the same fuel tank and oxidizer tank. Tanks are pressure stabilized, maintaining their structural rigidity without internal bracing.

The booster section comprises a booster engine system with two thrust chambers, an aft engine fairing assembly, a fireshield, thrust longerons to transmit the loads from the engines to the sustainer thrust ring, and the associated hydraulic, pneumatic, and electrical components required for booster engine operations. At the optimum time during the mission trajectory, the booster engines are cut off and the entire section is jettisoned to lighten the total vehicle weight.

A light-weight bulkhead separates the liquid oxygen compartment from the fuel tank in the sustainer section. Located on the centerline at the aft end of the vehicle is the main or sustainer engine. Mounted along the sides of the propellant tanks are the two small vernier motors. The sustainer is gimbal-mounted and provides directional control. After this engine is shut down, the verniers allow final correction in pitch, yaw and roll.

A radio inertial guidance system determines the vehicle's position and velocity and, on the basis of mission objectives, commands the flight correction necessary to fulfill these objectives. A pulse beacon receives a pulse-coded signal from a ground station containing discrete steering commands.

3.1.2.1.1 SLV-3 Configuration (Cont.)

The pulse beacon sends a return signal to the ground station for measurement of position of the vehicle. A decoder translates the pulse message from the ground station and distributes the information to the various vehicle systems. A rate beacon receives a double side-band suppressed carrier signal from the ground station and generates a return signal which lies between the frequencies of the two side-bands received.

3.1.2.1.2 Agena D Configuration

The Agena D spaceframe is a semimonocoque structure consisting of magnesium skin and supporting members providing the aërodynamic and structural shape of the basic vehicle. The spaceframe houses and supports the various vehicle systems and components, and provides the support platform for the spacecraft.

The Agena D is divided into four sections: Forward, propellant tank, aft, and booster adapter. The forward section contains mounting and supporting provisions for components of the pressurization, electrical, guidance, and electronics systems. The propellant tank consists of two compartments for separate storage of the fuel and oxidizer. The fuel is unsymmetrical dimethylhydrazine (UDMH), The oxidizer is inhibited red-fuming nitric acid (IRFNA). The aft section consists essentially of an engine cone and support structure for mounting the engine flight control pneumatic system, hydraulic system, and associated wiring harnesses and brackets. Equipment trays, for additional items necessary for a specific mission, are also placed in the aft section. The adapter section and extension mate the Agena with the SLV-3 booster.

3.1.2.1.2 Agena D Configuration (Cont.)

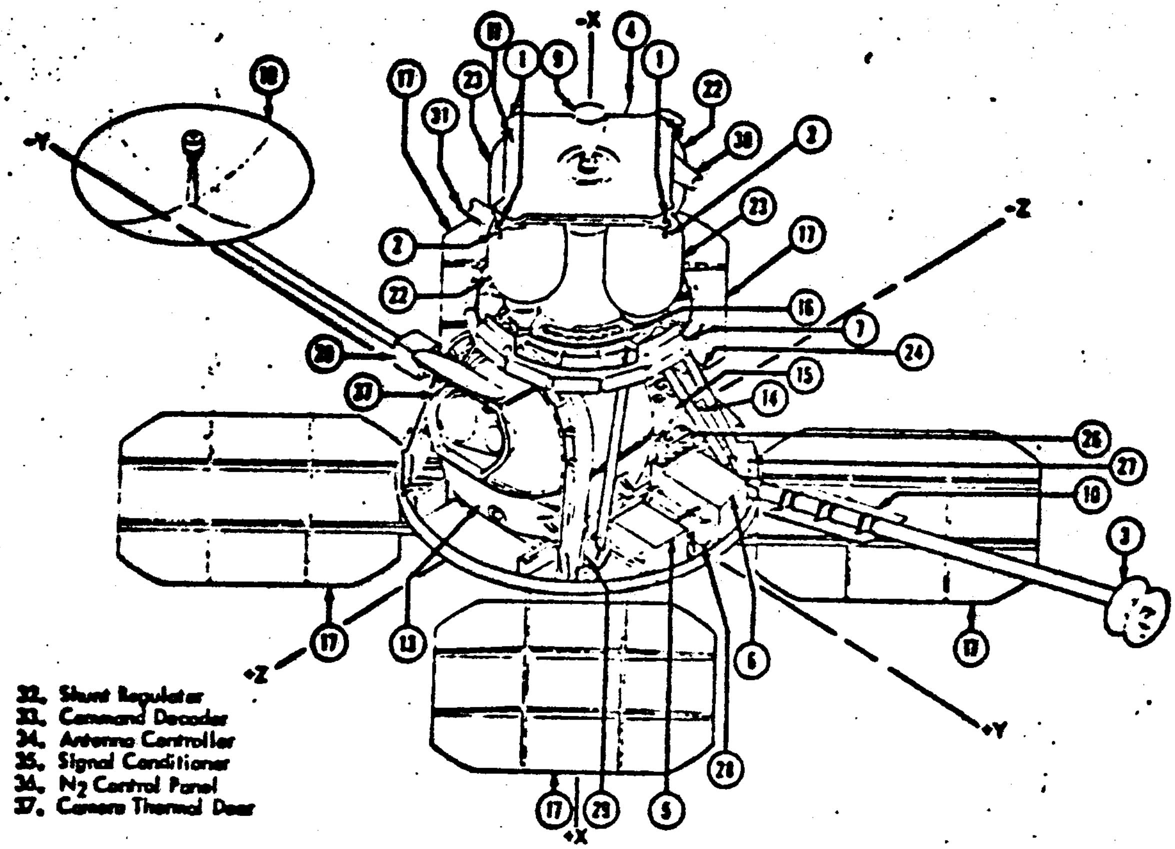
The adapter also houses two retrorocket motors which provide the impulse required to separate Agena from SLV-3.

The Agena D flight control system carries out vehicle stabilization commands from the guidance system and performs the required time sequencing functions after the SLV-3 is jettisoned.

During powered flight, pitch and yaw control are achieved by gimbaling the main engine thrust chamber through a controller on each of the hydraulic actuators. Roll control during powered flight, and pitch, yaw, and roll control during the coast periods is achieved by cold gas thrust valves using a mixture of nitrogen and freon.

Vehicle attitude is sensed by a three-axis, strapped-down, inertial-reference package that is referenced to the earth by an infrared horizon sensor system. Attitude errors are converted to the required corrective torques which are applied to the vehicle through the control system. Vehicle velocity is measured along the line-of-flight by an integrating accelerometer unit known as the velicity meter. During powered flight operations, the velocity meter generates the engine cutoff signal when the required velocity increment is reached. A C-band transponder is installed in the vehicle for tracking purposes.

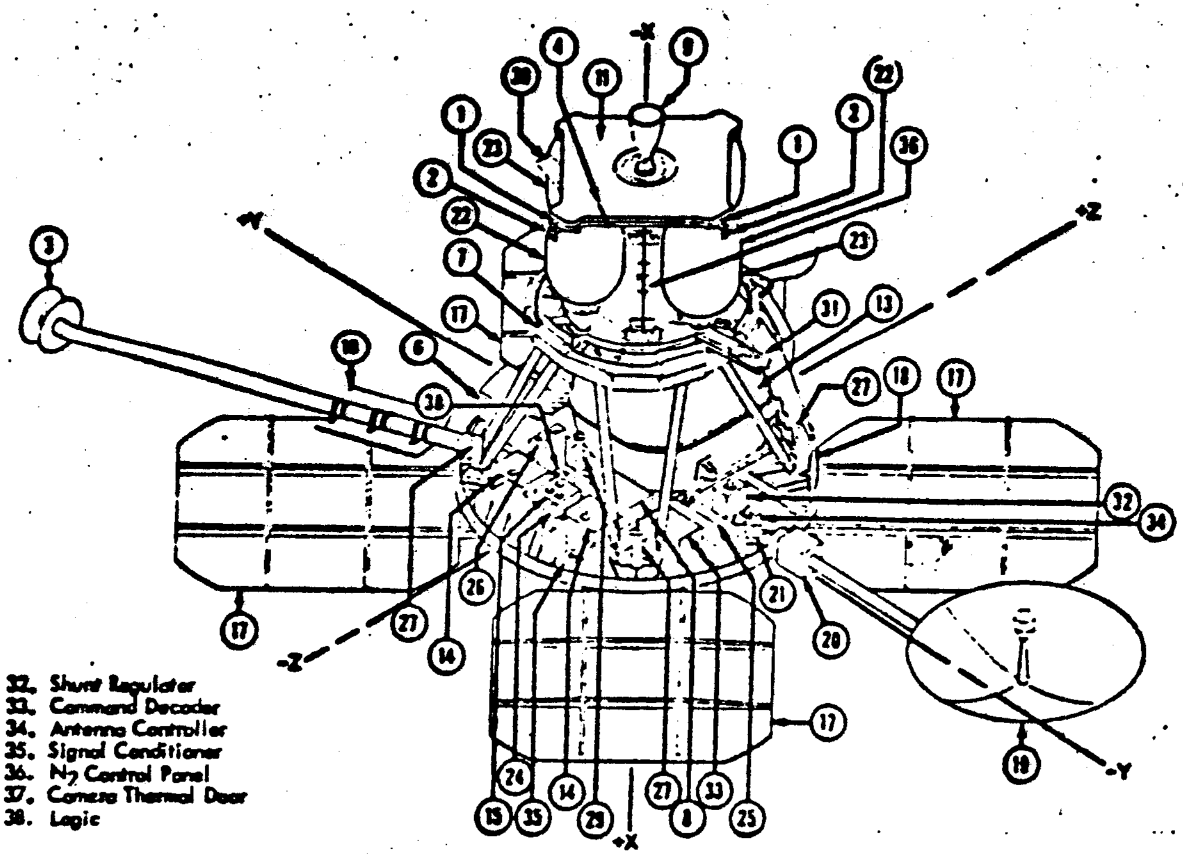
A telemetry system monitors and measures functional and environmental conditions in the Agena and transmits this data on a VHF carrier. In addition, one channel of the telemetry system will be utilized for transmission of the complete spacecraft data stream.



1. Reaction Control Thrusters
2. Coarse Sun Sensor
3. Low Gain Antenna
4. R-F Pickup
5. Inertial Ref. Unit
6. Control Assy. (Flt. Programmer)
7. Micrometeoroid Detector
8. Multiplexer Encoder
9. Velocity Control Engine
10. Shunt Reg. Pwr. Resistors
11. Heat Shield
12. Thermal Barrier (Not Shown)
13. Photographic Subsystem
14. Battery
15. Charge Controller
16. Shunt Reg. Emitter Resistors
17. Solar Panel
18. Shunt Reg. Pwr. Transistors
19. High Gain Antenna
20. High Gain Antenna Drive
21. High Gain Antenna Actuator
22. Fuel Tank
23. Oxidizer Tank
24. Transponder
25. TWTA
26. Modulation Selector
27. Solar Panel Actuator
28. Canopus Star Tracker
29. Scintillation Counter
30. Low Gain Ant. Support Bracket
31. High Gain Ant. Support Bracket
32. Shunt Regulator
33. Command Decoder
34. Antenna Controller
35. Signal Conditioner
36. N_2 Control Panel
37. Camera Thermal Door

SPACECRAFT CONFIGURATION

FIGURE 3.1.2-2



1. Reaction Control Thrusters
2. Coarse Sun Sensor
3. Low Gain Antenna
4. R-F Pickup
5. Inertial Ref. Unit
6. Control Assy. (Flt. Programmer)
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32. Shunt Regulator
33. Command Decoder
34. Antenna Controller
35. Signal Conditioner
36. N_2 Control Panel
37. Camera Thermal Door
38. Logic

FIGURE 3.1.2-3

SPACECRAFT CONFIGURATION

3.1.2.1.2 Agena D Configuration (Cont.)

Spacecraft data will be conditioned and supplied to the Agena telemetry system through an in-flight disconnect.

3.1.2.2 Spacecraft

The Lunar Orbiter spacecraft, opposite views of which are shown in Figures 3.1.2-2 and 3.1.2-3, has a nominal weight of 850 pounds and is designed to be mounted within an aerodynamic nose fairing on top of the Atlas/Agena launch vehicle.

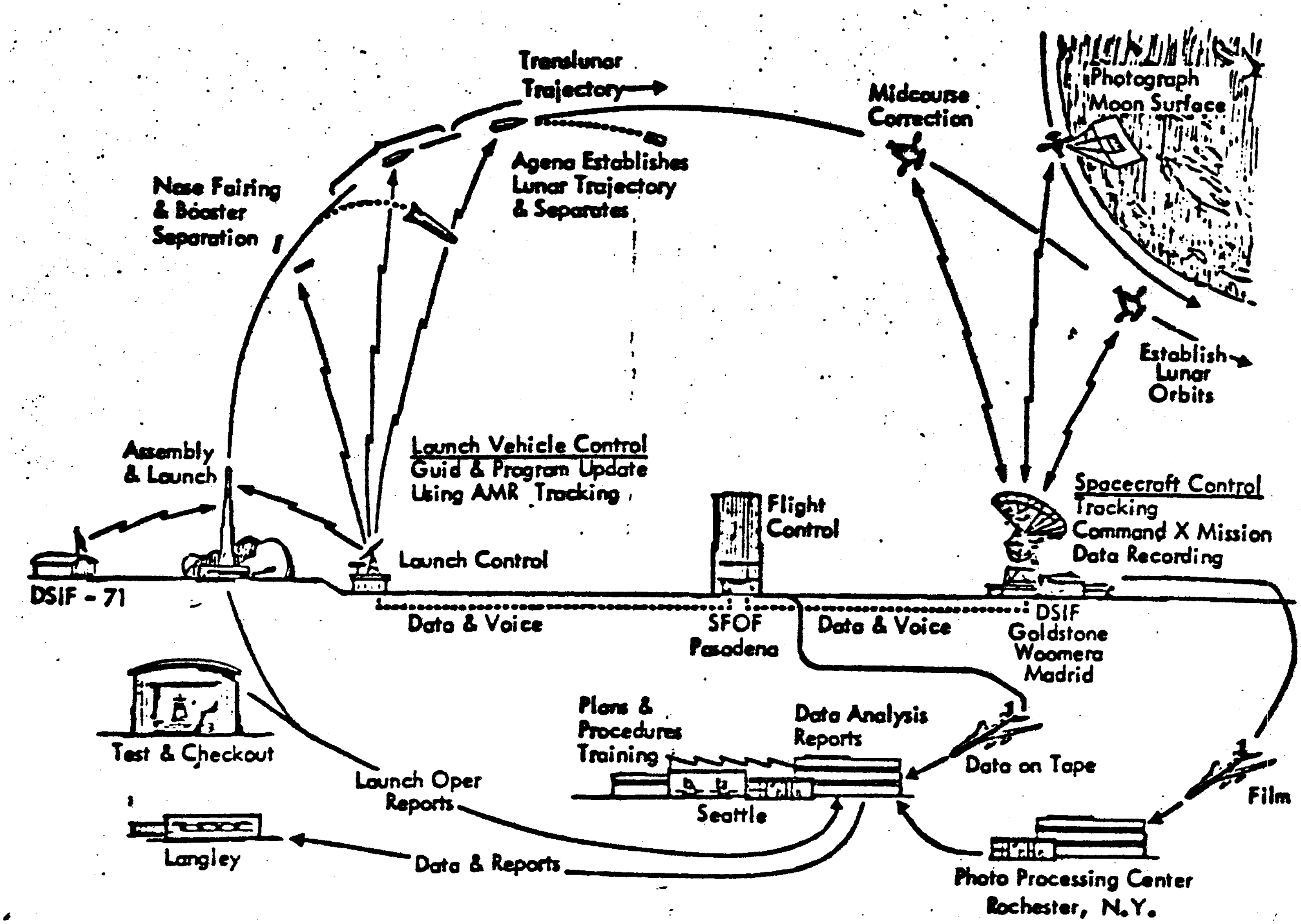
During launch, the solar panels are folded under the spacecraft base and the antennas are held against the side of the structure. In this configuration, the spacecraft is approximately 5 feet in diameter and 5½ feet high. With the solar panels and antennas deployed after injection into the translunar trajectory, the maximum span is increased to approximately 18½ feet along the antenna booms and 12 feet across the solar panels.

The spacecraft is composed of the following major elements:

a. Photo Subsystem
b. Power Subsystem
c. Communications Subsystem
d. Attitude Control Subsystem
e. Velocity Control Subsystem
f. Structures and Mechanisms Subsystem
g. Spacecraft Subsystem "Integrated Elements" (consists of those elements of hardware not specifically included in the subsystems a. through f., above).

3.1.2.3 Block I Mission Profile

After launch from Pad 13 at Cape Kennedy, mission control and data acquisition will be exercised from the Air Force Eastern Test Range (AFETR) until handover to the Deep Space Network (DSN) following spacecraft injection into translunar trajectory. The DSN is comprised of the Space Flight Operations Facility (SFOF),



TYPICAL MISSION OPERATIONS

FIGURE 3.1.2-4

3.1.2.3 Block I Mission Profile (Cont.)

the Deep Space Instrumentation Facility (DSIF), and a Ground Communications System (GCS). Deep Space Stations (DSS's) within the DSIF will be used for the transmission of SFOF-originated commands to the spacecraft and for the acquisition of spacecraft data for use at the SFOF. The transfer of control from the AFETR to the DSN will take place upon positive acquisition of a spacecraft by a DSS. Figure 3.1.2-4 shows typical operations for the complete mission.

3.1.2.3.1 AFETR Mission Phase

Due to lunar illumination requirements and waiting time assumptions, the launch period for one mission, identified as F-I, is from June 16, 1966 to June 23, 1966. June 20 and June 21 will likely be unavailable, however, because of Agena Horizon Sensor constraints. During this launch period, the available launch window on the AFETR will be greater than the 15 minute minimum. The specific F-1 trajectory used for planning purposes includes liftoff at (GMT) on June 23, 1966. The boost trajectories for Lunar Orbiter include an earth parking orbit with coast times varying with lunar declination and launch azimuth.

Upon launch, the vehicle will rise vertically and execute a programmed roll maneuver so that the proper launch azimuth (as determined by the launch time) will be obtained when a programmed pitch-down maneuver is initiated. This programmed pitch-down maneuver will be performed until the Atlas booster engines are cut off and jettisoned. During the subsequent sustainer and vernier stages, adjustments in vehicle attitude and engine cut-off times will be commanded to adjust the position and velocity at Atlas vernier engine cut-off. Immediately following vernier engine cut-off, the nose fairing will

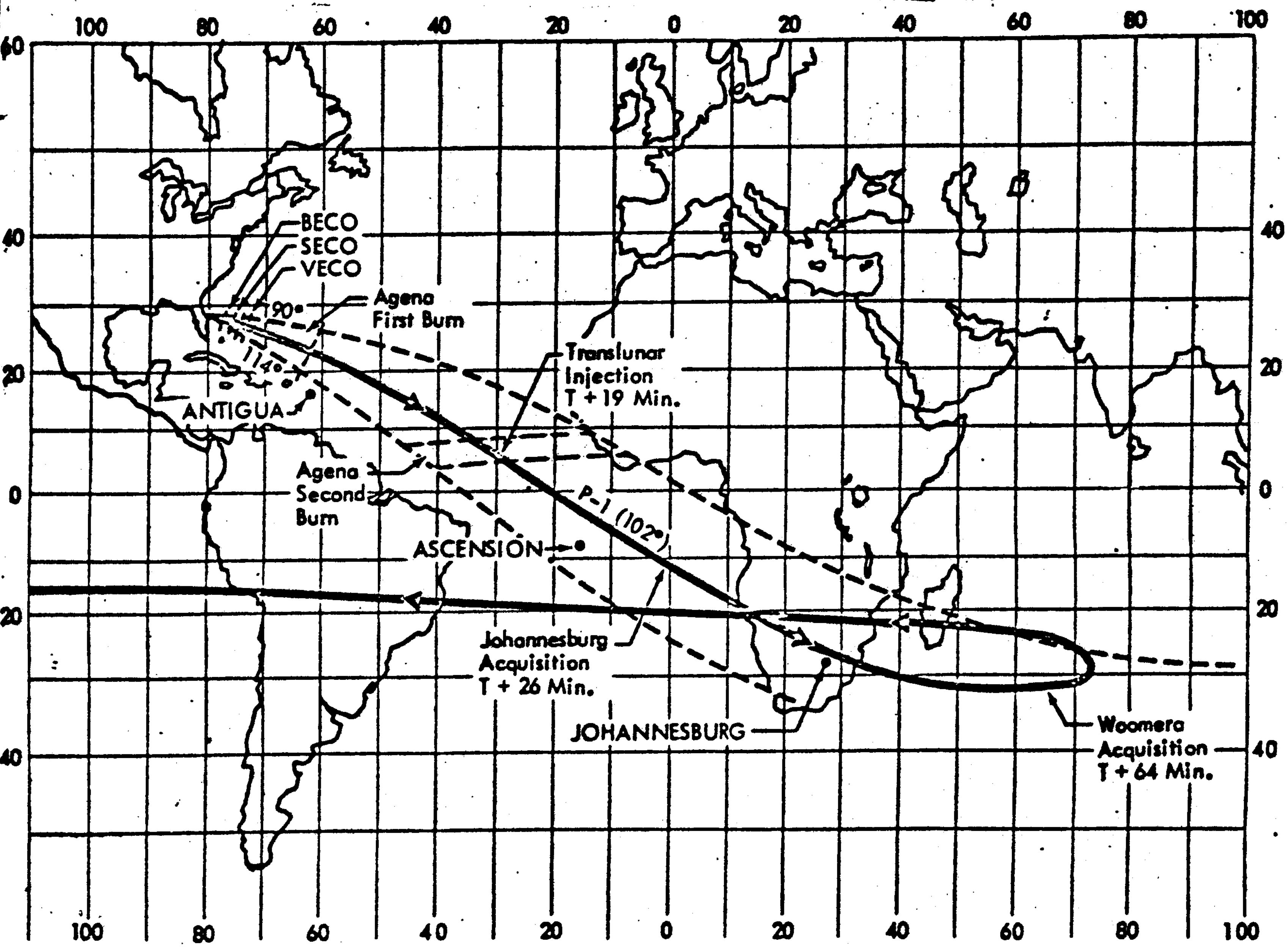
3.1.2.3.1 AFETR Mission Phase (Cont.)

be ejected and Atlas-Agena separation will occur. Agena first-burn will then be executed to inject the vehicle into a 100-nautical-mile parking orbit. The first-burn maneuver will be programmed pitch-down maneuver.

At a predetermined time in earth parking orbit, Agena second-burn will be initiated to inject the vehicle into the translunar trajectory. This will be followed by the separation of the spacecraft from the Agena and by an Agena retromaneuver to reduce the probability of Agena interference with the spacecraft, or Agena impact on the moon. Figure 3.1.2-5 shows the initial portion of the trajectory through injection into the translunar trajectory.

Prior to launch, the Cape Kennedy DSS will provide the SFOF with detailed information, as established by a mutually agreed upon countdown, such as spacecraft RF link characteristics, to assist in subsequent spacecraft acquisition by the DSIF. From Launch until transfer of mission control to the SFOF, AFETR will provide the SFOF with tracking data and spacecraft telemetry data in real time. This data, which will be processed at Cape Kennedy, will be acquired from the AFETR downrange C-band tracking stations and VHF and certain S-band telemetry stations.

Tracking and pertinent Agena telemetry-events data will be used by AFETR to determine the parking orbit and transfer orbit conditions. The computed data will then be forwarded to the Cape Kennedy JPL Operations Center for relay to the SFOF. A portion of the data used by AFETR will also be relayed to the SFOF where it will be used in the JPL Orbit Determination process. In addition, initial



INITIAL P-1 TRAJECTORY

FIGURE 3.1.2-5

3.1.2.3.1 AFETR Mission Phase (Cont)

acquisition data (Look Angles) for Johannesburg and Woomera DSS's will be provided. This data will be computed and relayed to the SFOF for evaluation and onward transmission to the DSIF.

During the AFETR mission phase, spacecraft performance telemetry data will be transmitted by the spacecraft S-band transmitter. The same information will also be contained on the Agena VHF telemetry system, from launch through Agena-Spacecraft separation. Data from both sources received by the downrange stations will be recorded locally and data from at least one source will be transmitted to the SFOF via Cape Kennedy. Downrange station recordings will be made available to the Project for evaluation.

3.1.2.3.2 DSN Mission Phase

From acquisition of the spacecraft by the DSIF until the end of the mission, control will be exercised from the DSN. Three deep Space Stations (DSS's), Goldstone-Echo, Madrid, and Woomera, will be used to transmit commands to the spacecraft and obtain angular position, doppler, ranging, photo data, and performance telemetry data. In addition, the Johannesburg DSS may be required to track the spacecraft and record performance telemetry data during the initial phase of the translunar trajectory. All commands will either be originated at or approved by the SFOF.

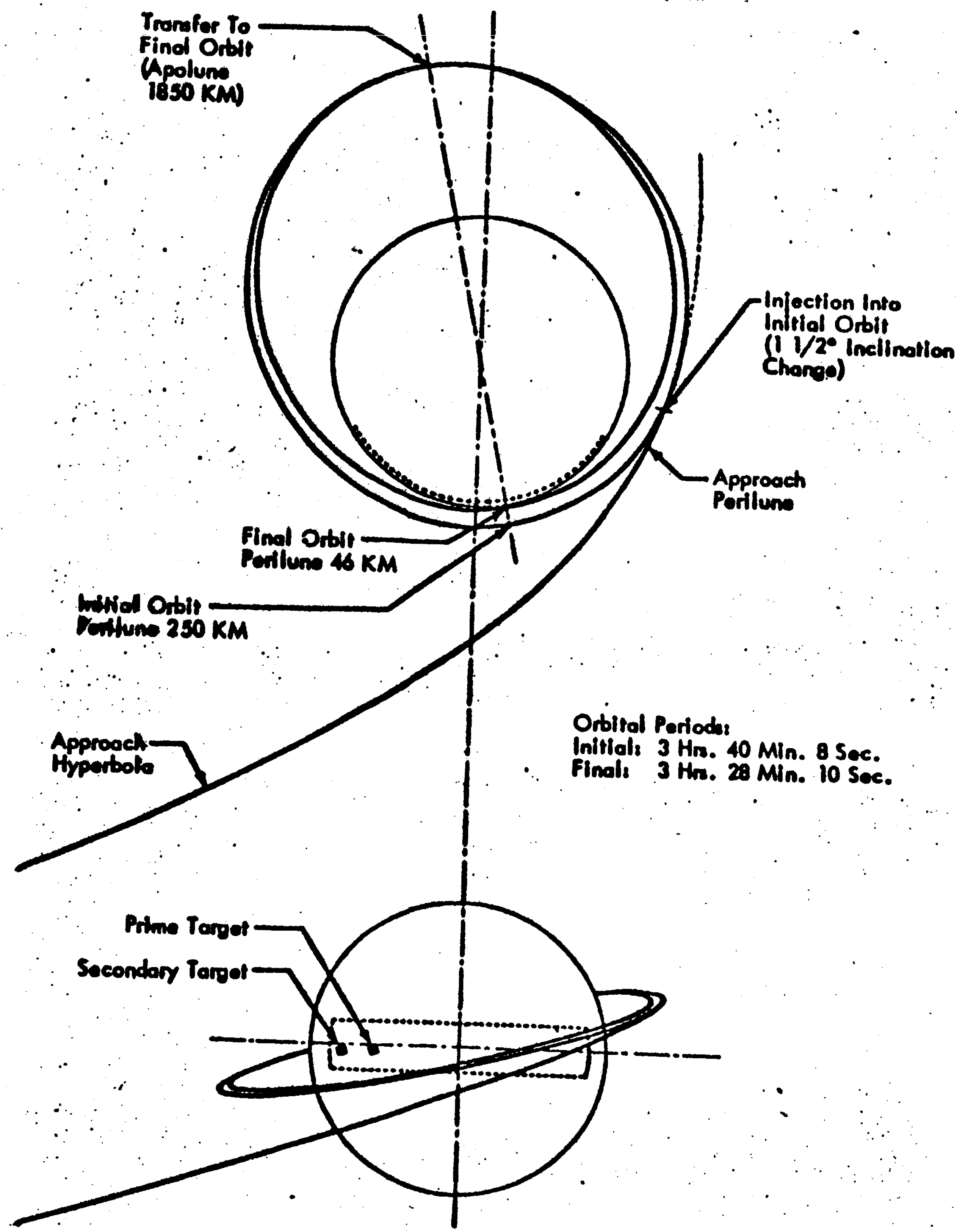
Shortly after Agena-Spacecraft separation, the spacecraft programmer will command solar panel and antenna deployment and sun acquisition. Upon passing through the Van Allen belt, after about six (6) hours of translunar flight, the Canopus sensor will be turned on. Star mapping will then be accomplished to facilitate Canopus identi-

3.1.2.3.2 DSN Mission Phase (Cont.)

fication and acquisition. Spacecraft attitude will be established and maintained throughout the mission by the reaction control system working in conjunction with the Canopus and sun sensors. Based on tracking data obtained through the DSIF, a trajectory course correction maneuver will be commanded by the SFOF after about 15 hours of translunar flight, if required. If necessary, a correction may be made as early as seven(7) hours. After approximately 70 hours of translunar flight, another course correction will be commanded if tracking data indicates the need.

As the spacecraft approaches the moon, an SFOF-originated, lunar-orbit injection command will be executed to place the spacecraft into an initial orbit with a 250 km perilune and an apolune of 1850 KM. During the initial orbit phase, which will have a duration of about 45 hours minimum, 17 photo frames will be taken and 3 read out to permit identification of subsystem optimization adjustments which may be necessary prior to photography in the final orbit phase. Toward the end of the initial orbit phase, a transfer command will be loaded into the spacecraft programmer. Execution of this command will place the spacecraft in a final orbit with a 46 km perilune and an apolune of 1850 KM. The geometry of the initial and final orbits is shown in Figure 3.1.2-6.

In the final orbits, the spacecraft will be commanded to take 11 photo sequences; 7 of the prime target and 4 of the secondary target. Between photo orbits, limited readout will be commanded to verify proper operation of the photo subsystem. Readout of all photo frames will commence on the 27th final orbit. This will continue during view periods, throughout the 30-day mission, with the



ORBITAL GEOMETRY (INJECTION CONDITIONS)

FIGURE 3.1.2-6

3.1.2.3.2 DSN Mission Phase (Cont.)

exception of every 9th orbit, during which readout commands will be loaded in the spacecraft programmer and tracking data acquired for orbit determination.

During the initial 30 days of lunar orbit, all film (Up to 194 pictures) will be exposed, developed, and transmitted to earth. Vehicle stabilization will be maintained to a closer tolerance during maneuvers, photography and video data transmission than at other times.

Until completion of the photo phase, the DSN will continuously track the spacecraft from acquisition through that portion of the lunar orbit where it is not obscured by the moon. Thereafter, for a period of 11 months, the spacecraft will be tracked periodically to obtain scientific data. Tracking requirements are as follows: (1) For a period of 30 days beginning at end of photo transmission: 1-minute sampling over four orbits per day by any tracking station; (2) from 30 days after end of photo transmission to one year after launch: 1-minute sampling of two orbits per day, three days per week, from any tracking station. A maximum of two to three days separation between these samples is preferred.

3.2 Subsystem Descriptions

3.2.1 Photo Subsystem

3.2.1.1 Introduction

The Photographic Subsystem of the Lunar Orbiter has been designed to obtain both high and medium resolution photographs of the lunar surface. The high resolution coverage is centered within the area of medium resolution. Photographs nominally will be obtained from

3.2.1.1 Introduction (Cont.)

the spacecraft while in an elliptical orbit which places the perilune over the target area at an altitude of 46 kilometers. Operational modes are provided to permit different coverage patterns of high and medium resolution, including overlap suitable for stereo. Photographs are taken on film which is processed and scanned to produce a video signal for transmission to Earth, where they are reconstructed.

3.2.1.2 Functional Description

The Photographic Subsystem (PS) has several basic units having distinct functions. These include the camera, processor, and readout, and the associated power supply and electronics in the spacecraft, and the Ground Reconstruction Equipment (GRE). A schematic diagram of the Lunar Orbiter Photographic Subsystem is shown in Figure 3.2.1-1.

3.2.1.2.1 Camera

The photographs are taken by a dual-lens system which produces a high resolution image and a medium resolution image on 70 mm film. The medium resolution lens is a Schneider Zenotar of 80 mm focal length which operates at a fixed Aperture of f:5.6. The high resolution lens is a Paxoremic, having a focal length of 24 inches and the same aperture of f:5.6. A between-the-lens shutter is used by the 80 mm lens, and a double-curtain focal plane shutter by the 24-inch lens. These shutters are adjustable by ground command to 1/25, 1/50 or 1/100 second. The film is held in the focal planes by platens which clamp the film and hold it flat to ± 0.001 inch by vacuum during exposure.

PHOTO SUBSYSTEM SCHEMATIC

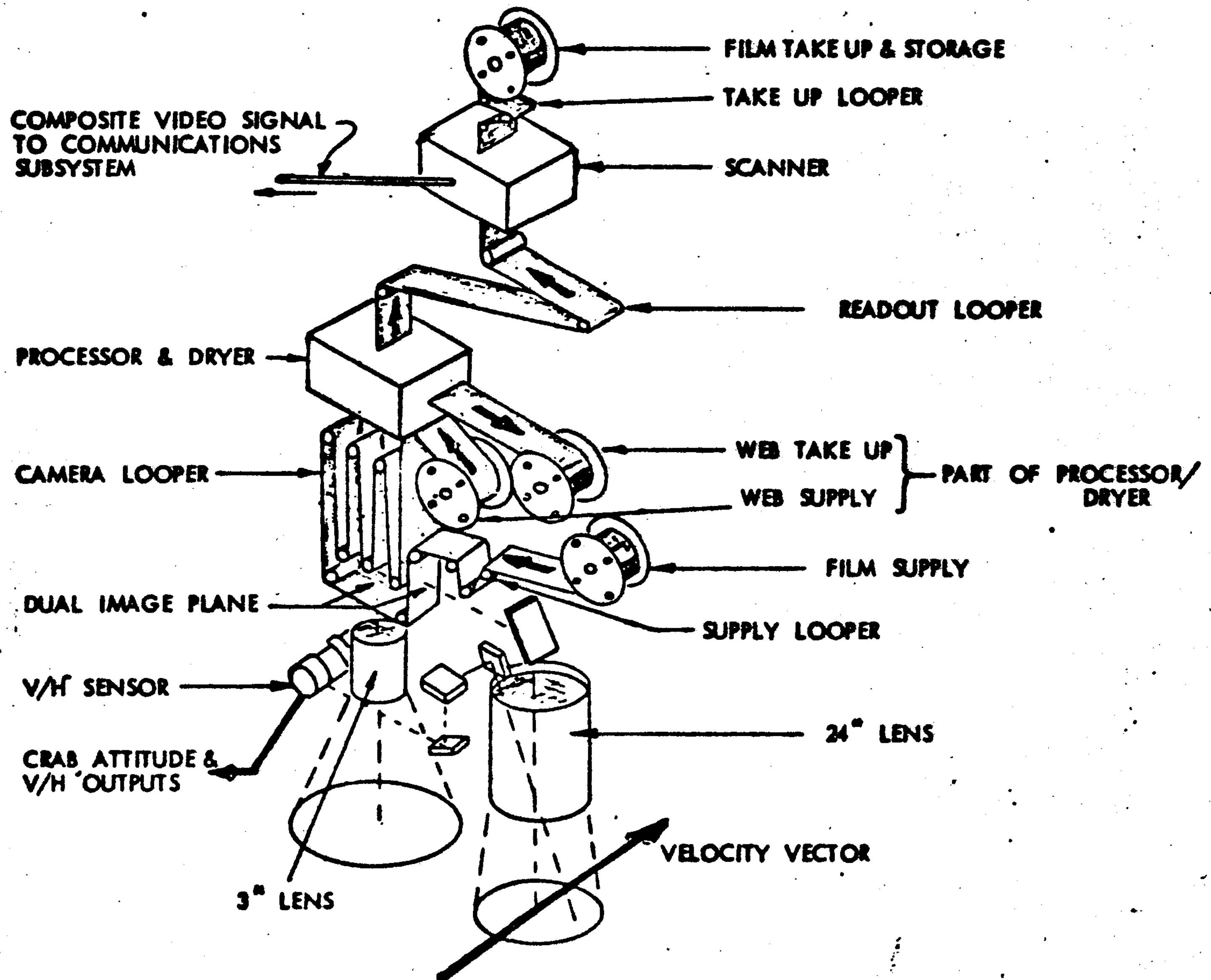


FIGURE 3.2.1-1

3.2.1.2.1 Camera (Cont.)

Because of orbital velocity, image motion compensation is required to prevent image smear. A velocity-to-height ratio (V/H) sensor tracks a portion of the image produced by the 24-inch lens, and translates tracking motion in its optical train to a platen movement, which follows the image along the direction of flight. Both platens are moved. A 1 σ error of about 2 microns during a 1/50 second exposure has been calculated for nominal image smear over the altitude range of 38 to 210 kilometers. An illumination level of 20-1000 foot lamberts at the surface is required.

At the nominal perilune altitude of 46 kilometers, resolution of 1 meter on the surface will be obtained in the high resolution photographs, and 8 meters resolution in the medium resolution photographs with an object contrast ratio of 3:1.

The film used on the Lunar Orbiter missions is Eastman-Kodak Type SO-243. This film has an aerial exposure index of 1.6. Although slow in comparison with more common emulsions, it has extremely fine grain and exceptionally high resolving power. In addition, it has a low sensitivity to ionizing radiation. To provide for control and calibration, data consisting of high and low contrast resolution bars, a ten-step gray scale, and linearity pattern, as shown in Figure 3.2.1-2, is pre-exposed along one edge of the film under precisely controlled conditions. A total of 260 feet of film, sufficient for 194 dual exposure frames, is carried in the Photographic subsystem.

The picture formats, dictated by film width, lens field of view and the requirements of photographic coverage and resolution are

PRE-EXPOSED EDGE DATA

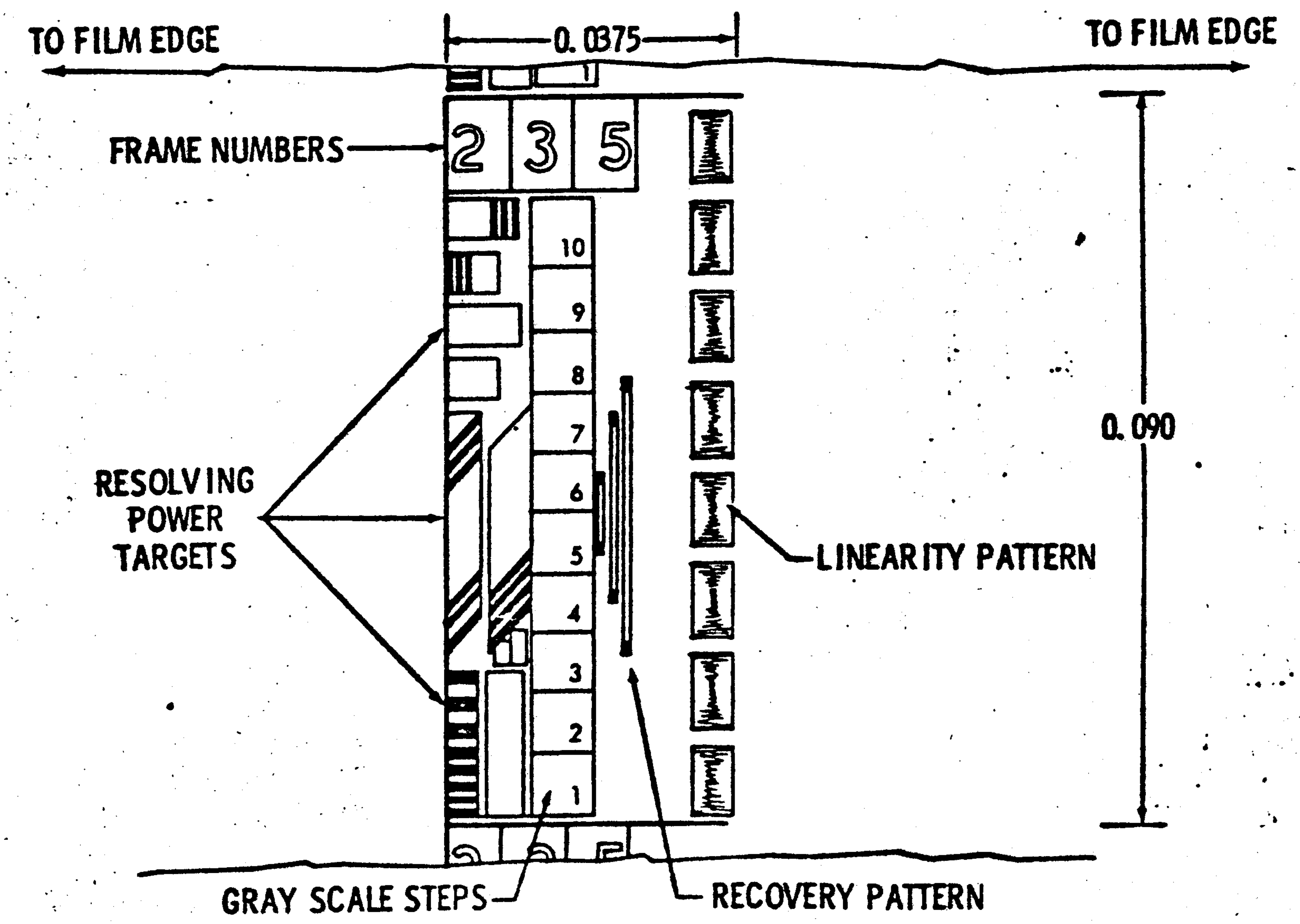


FIGURE 3.2.1-2

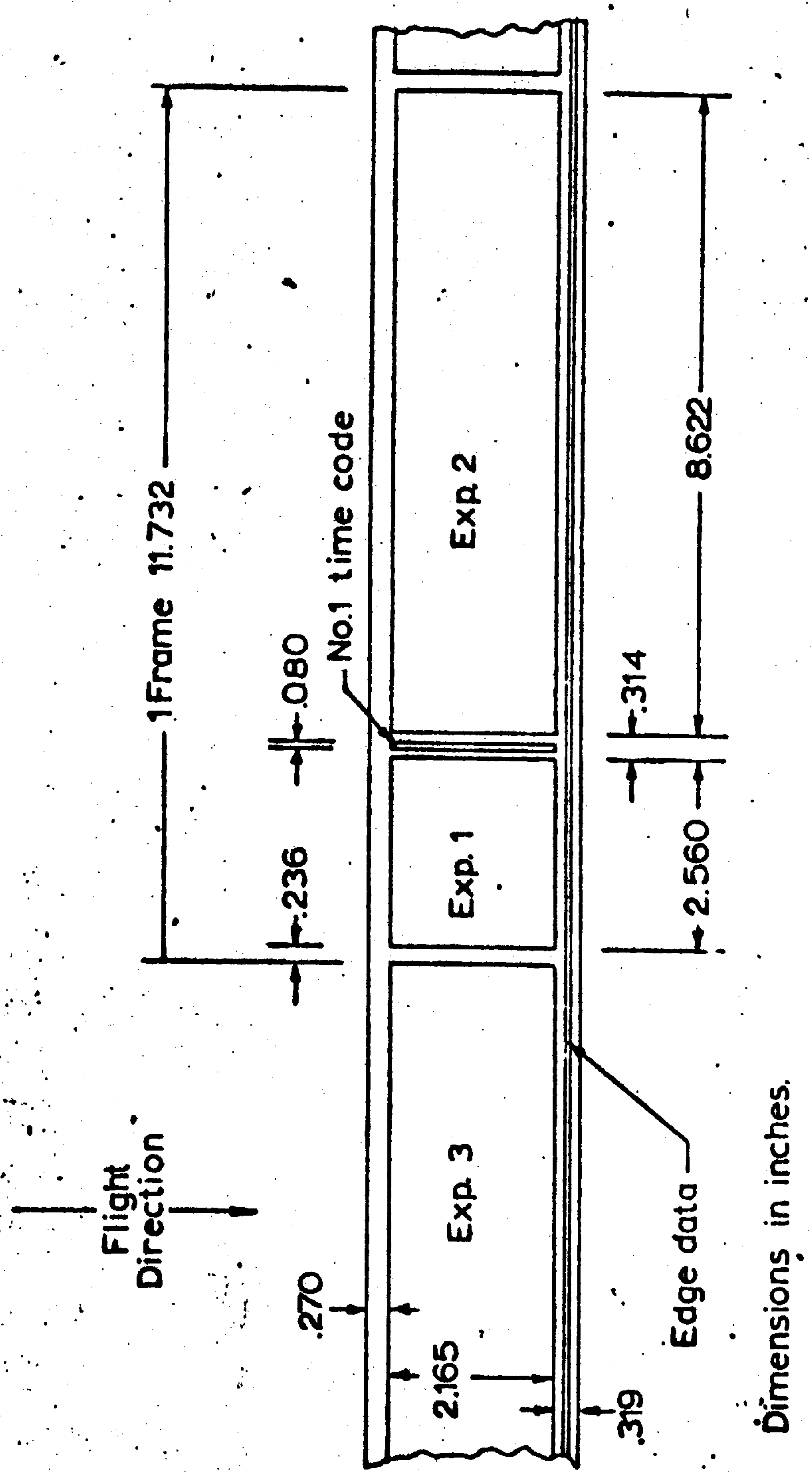
3.2.1.2.1 Camera (Cont.)

shown in Figure 3.2.1-3. The format for the high resolution photographs is 55 x 219 mm corresponding to half-angles of 3^o and 10^o. For the medium resolution photographs the format is 55 x 65 mm, or 19^o x 22^o half-angles. The direction of flight is across the narrower dimension. The equivalent coverage on the lunar surface is dependent upon the altitude of the spacecraft and to any deviation of the camera optical axis from the spacecraft nadir. At the nominal perilune altitude of 46 kilometers, the high and medium resolution photographs cover areas of 4.15 x 16.6 kilometers and 31.6 x 37 kilometers respectively. Both exposures are made simultaneously. Because of lens separation and the use of a folding mirror in the 24-inch lens optical path which orients the focal planes at an angle of 90^o, the images of each lens for a particular photo pair are not adjacent on the film.

Since photographs can be sequenced as single-frame exposures, or as a series of 4, 8, or 16 frames at two rates to provide the different coverage patterns, film will be exposed at a rate much faster than it can be processed. A looper having a capacity of 20 frames takes up the film and holds it until processed.

3.2.1.2.2 Processor - Dryer

The exposed film is processed at a rate of 2.3 inches per minute by the Eastman-Kodak "Bimat" process. The exposed film is laminated with the Bimat film which carries a single developer-fixer solution absorbed in its emulsion layer. Processing occurs by diffusion of the reagent from the Bimat to the film. Since the process goes to completion, lamination of the Bimat and film may vary from a minimum



70mm. FILM EXPOSURE FORMAT

FIGURE 3.2.1-3

3.2.1.2.2 Processor - Dryer - (Cont.)

of three minutes to a maximum of several hours. Following delamination from the Bimat, the slightly damp film passes through a dryer. Moisture is taken up by pads containing potassium thiocyanide (KSCN) which maintain a 50% relative humidity within the enclosed pressurized shell containing the Photo Subsystem. The dried film is then stored on a take-up reel until read out.

3.2.1.2.3 Readout (Optical Mechanical Scanner)

During readout, an electron gun sweeps a line of light across a phosphor covered rotating drum in the Line Scan Tube (LST). An optical system focuses an image of the line on the photographic film, resulting in a scan line 2.67 mm in length. The optical system mechanically indexes the position of the line to produce the sweep in the orthoganal direction. When scanning has progressed across the width of the film, the film is indexed ahead by an amount equal to the length of the sweep to scan the next "framelet". The light transmitted through the film, modulated by the image density, is sensed by a photomultiplier tube which generates a signal proportional to the intensity of the transmitted light. This signal is amplified, timing and synchronization pulses added, and fed into the communications link as the composite video signal for transmission to Earth. Figure 3.2.1-4 shows the important features of the readout.

As film motion through the scanner during readout must be opposite to that during photography, a readout looper is provided between the processor-dryer and the scanner to take up a length of film adequate to read out three dual-frames before all film has been exposed and processed since processing cannot be reversed. Following com-

READOUT

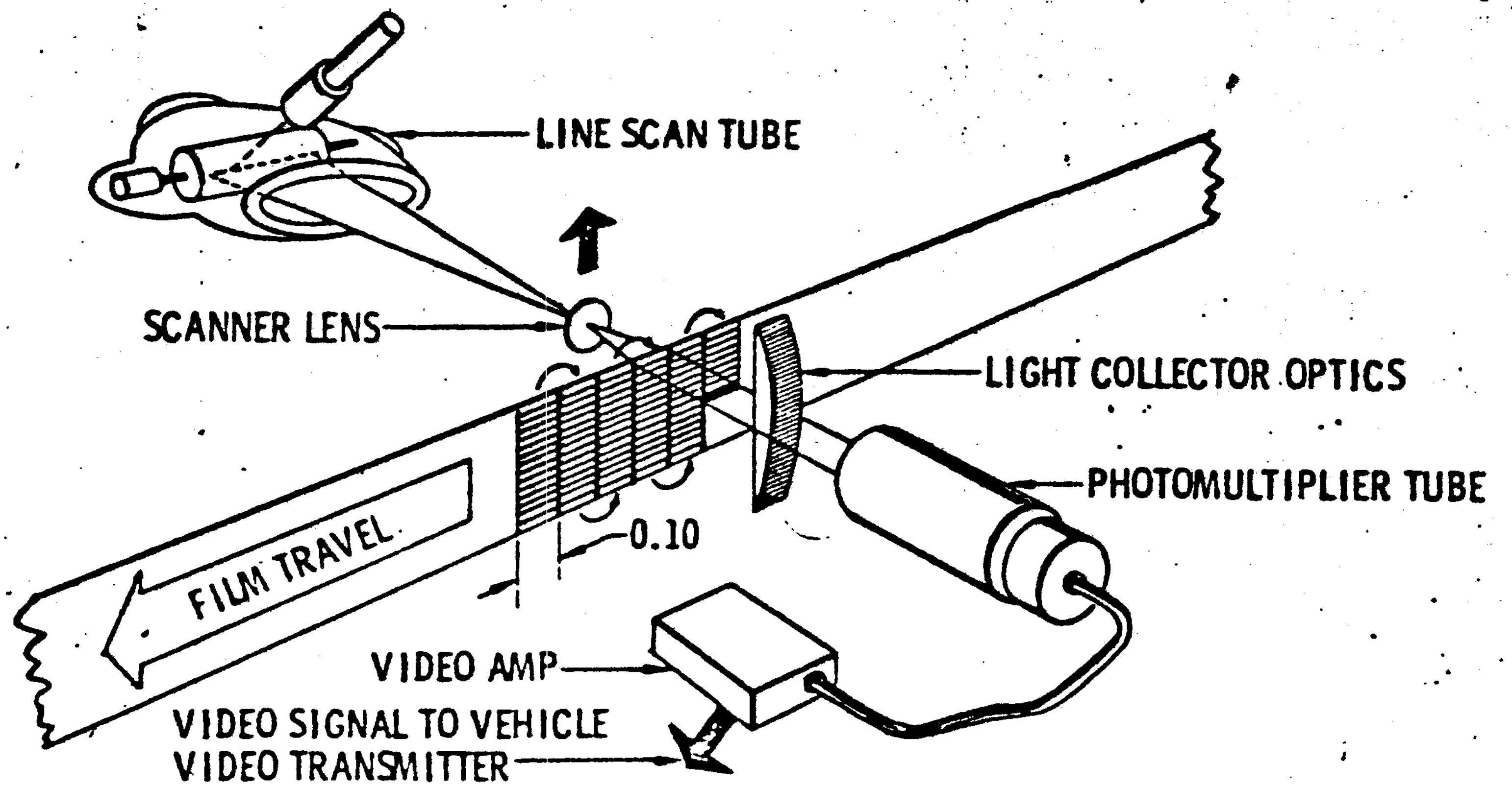


FIGURE 3.2.1-4

3.2.1.2.3 Readout (Optical Mechanical Scanner) (Cont.)

pletion of all photography, the Bimat is cut, allowing film to be run back to the original supply reel while scanning all of the photographs. Provision is made for repeated readout if desired.

3.2.1.2.4 Command, Control and Programmer

The Command, Control, and Programmer has the function of providing the programmed operating logic, sequencing, verification of programmed or real time commands, and handling of most PS telemetry signals. The command list for PS operation is given in Table 3.2.1-1 and the telemetry for monitoring operations of the PS is given in Table 3.2.1-2.

3.2.1.2.5 Ground Reconstruction Equipment

The signal transmitted by the spacecraft is received on the earth by DSIF. The video portion is reconverted to an image on a cathode ray tube which is photographed by the recording kinescope on 35 mm film. The sections of film corresponding to each framelet are re-assembled in proper sequence and orientation by the re-assembly printer and recorded on 9 inch film. This sequence is shown in Figure 3.2.1-5.

3.2.1.3 Operational Description

The operation of the PS is determined by the mission requirements of target coverage including area, overlap requirements and resolution, and the constraints imposed by illumination requirements, target location, orbit parameters, film capacity, and time. For a given target location, photography is limited by the requirement of illumination at phase angles between 50° and 75° for suitable contrasts, and illumination levels within the exposure requirement dictated by the shutter speeds of 1/25, 1/50 and 1/100 second. Area coverage and overlap is governed by the exposure sequencing rate, the number of exposures made during passage over the target area, inclination

TABLE 3.2.1-1

PHOTO SUBSYSTEM COMMAND LIST

SPC 1	Stored, programmed	V/H	On
SPC 2	Stored, programmed	Camera	On
SPC 3	Stored, programmed	Readout Electronics	On
SPC 4	Real Time	V/H	Off
RTC 5	Real Time	Readout Drive On	On
RTC 6	Real Time	Readout Drive Off	Off
RTC 7	Real Time	Camera Shutter Adv.	Advance one step
RTC 8	Real Time	Cut Bimat On	On
RTC 9	Real Time	Horizontal Size, Inc.	Increase one step
RTC 10	Real Time	Horizontal Size Dec.	Decrease one step
RTC 11	Real Time	LST Focus, Inc.	Increase one step
RTC 12	Real Time	LST Focus, Dec.	Decrease one step
RTC 13	Real Time	Photovideo Gain, Inc.	Increase one step
RTC 14	Real Time	Photovideo Gain, Dec.	Decrease one step
RTC 16	Real Time	Wind Forward On	On
SPC 18	Stored, programmed	Solar Eclipse On	On
SPC 19	Stored, programmed	Solar Eclipse Off	Off
RTC 20	Real Time	Horizontal Center Inc.	Increase
RTC 21	Real Time	Horizontal Center Dec.	Decrease
SPC 26	Stored, programmed	Fast Camera Rate	Fast Rep. Rate
SPC 27	Stored, programmed	Slow Camera Rate	Slow Rep. Rate
SPC 28	Stored, programmed	Frame Count A	
SPC 29	Stored, programmed	Frame Count Not A	Sets Camera
SPC 30	Stored, programmed	Frame Count B	Program
SPC 31	Stored, programmed	Frame Count Not B	

TABLE 3.2.1-2

PHOTO SUBSYSTEM TELEMETRY LIST

PB 01	Take-up Real Contents	0-260 ft.
PB 02	Camera Storage Looper Contents	0-21 ft.
PB 03	Readout Looper Contents	0-48 inches
PB 04	Shutter Operations, 24-inch	0-31 counts
PB 05	Platen Operations	0-31 counts
PC 01	V/H ON/OFF	ON/OFF
PC 02	Camera ON	ON/OFF
PC 03	Camera Program Setting	0-7 counts
PC 04	Camera Shutter Setting	0-3 counts
PC 07	Bimat Clear	ON/OFF
PC 08	Bimat Take-up	ON/OFF
PC 09	Bimat Cut	ON/OFF
PC 10	Readout Electronics ON or Wind Forward ON	ON/OFF
PC 11	Readout Drive ON/OFF	ON/OFF
PC 12	LST Focus INC/DEC, or Photo Video Gain INC/DEC	ON/OFF
PC 15	Shutter Setting ADVANCE or Camera Rate FAST/SLOW	ON/OFF
PC 18	Camera Thermal Door OPEN	ON/OFF
PC 19	Camera Thermal Door CLOSED	ON/OFF
PE 01	10 Volt DC Converter Output	0-11 VDC
PE 03	LST Cathode Current	0-25 Micro Amp
PE 04	Video Output Voltage	0-5 VDC
PE 05	High Voltage Supply	0-21 K Volts
PE 06	Photomultiplier Voltage Supply	0-1100 Volts
PH 01	Lower PS Relative Humidity	40-60%
PH 02	Upper PS Relative Humidity	40-60%
PP 01	PS Pressure	0-5 psia
PR 01	V/H Ratio	0-0.06 rps
PT 01	V/H Sensor Temperature	40°-140°F
PT 02	Camera Temperature	40°-100°F
PT 03	Window Temperature	40°-100°F
PT 04	Processor Temperature	40°-100°F
PT 05	Dryer Temperature	40°-140°F
PT 06	Readout-Thermal Fin Plate Temperature	30°-130°F
PT 07	PS Environment Temperature, Upper	40°-140°F
PT 08	PS Environment Temperature, Lower	40°-140°F
PT 09	Upper Shell Temperature	40°-100°F

READOUT, RECONSTRUCTION & REASSEMBLY FORMATS

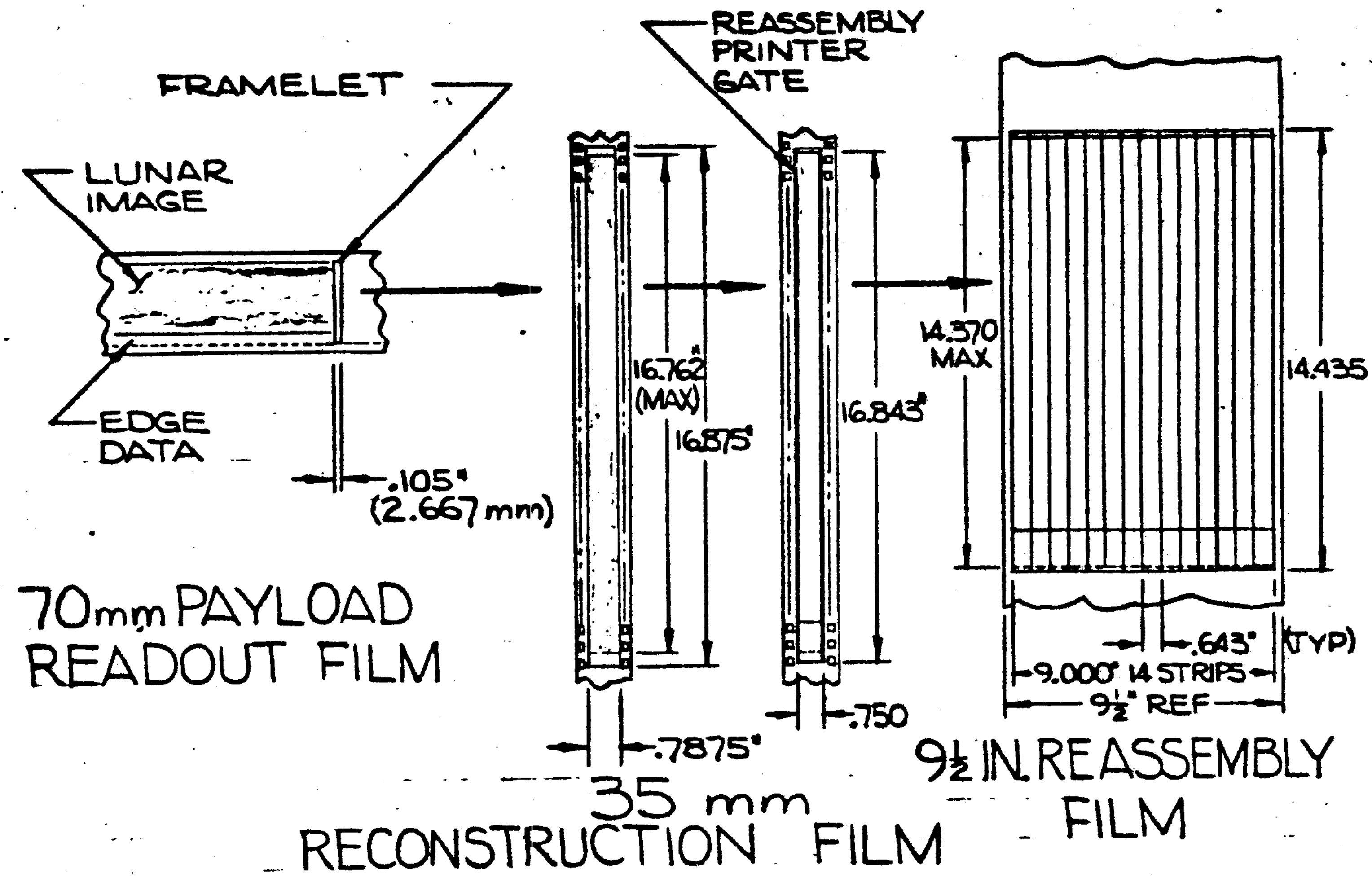
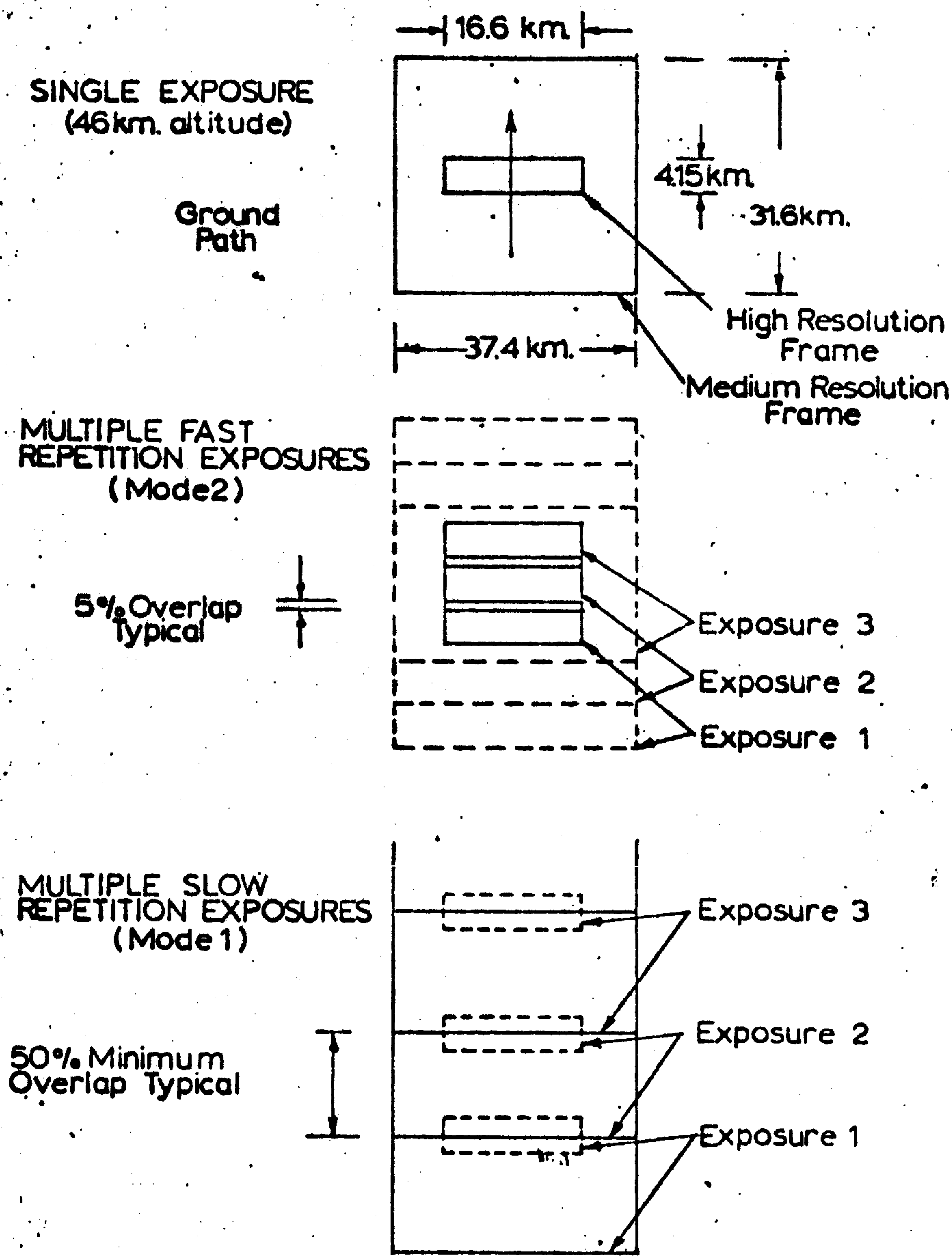


FIGURE 3.2.1-5

3.2.1.3 Operation Description (Cont.)

of the orbit, latitude of the target, and the number of orbits allowed for precession of the orbit nodes. The nominal photographic coverage for a single orbital pass is shown in Figure 3.2.1-6.

Photography, processing, and readout must be scheduled for those periods that the spacecraft solar panels are illuminated because of the limitations on battery power. In addition, readout is limited by the requirement that Earth cannot be occulted by the Moon during operation of the communications link.



NOMINAL LUNAR SURFACE PHOTOGRAPHIC COVERAGE

FIGURE 3.2.1-6

3.2.2 Power Subsystem

3.2.2.1 Subsystem Functional Description

Electrical power for the Lunar Orbiter is provided by a solar panel-battery system. During daytime the solar panels convert sufficient solar power to supply the spacecraft loads, the power subsystem losses, and charge the battery. During night time, and whenever the panels are oriented away from the sun, the loads are supplied from the battery. A simplified block diagram of the subsystem is shown in Figure 3.2.2.1.

3.2.2.1.1 Orbital Daytime Mode

When the solar array is illuminated, the maximum voltage at the outputs bus is limited by diverting excess power into dissipative elements associated with the shunt regulator. Sufficient power is provided to the charge controller to enable it to charge the battery at the maximum rate allowed by the battery temperature and state of charge. Minimum voltage to all loads is determined by the solar array voltage.

3.2.2.1.2 Orbital Night-time Mode

During orbital night-time the solar array, charge controller and shunt regulator are inactive and the power output voltage is determined by the battery voltage.

3.2.2.1.3 Power Distribution

Power for the actuation of the squibs is taken directly from the battery terminals without going through the battery diode. All other spacecraft loads are supplied through separate pins on the power output connector.

3.2.2.2 Component Descriptions

POWER SUBSYSTEM, BLOCK DIAGRAM

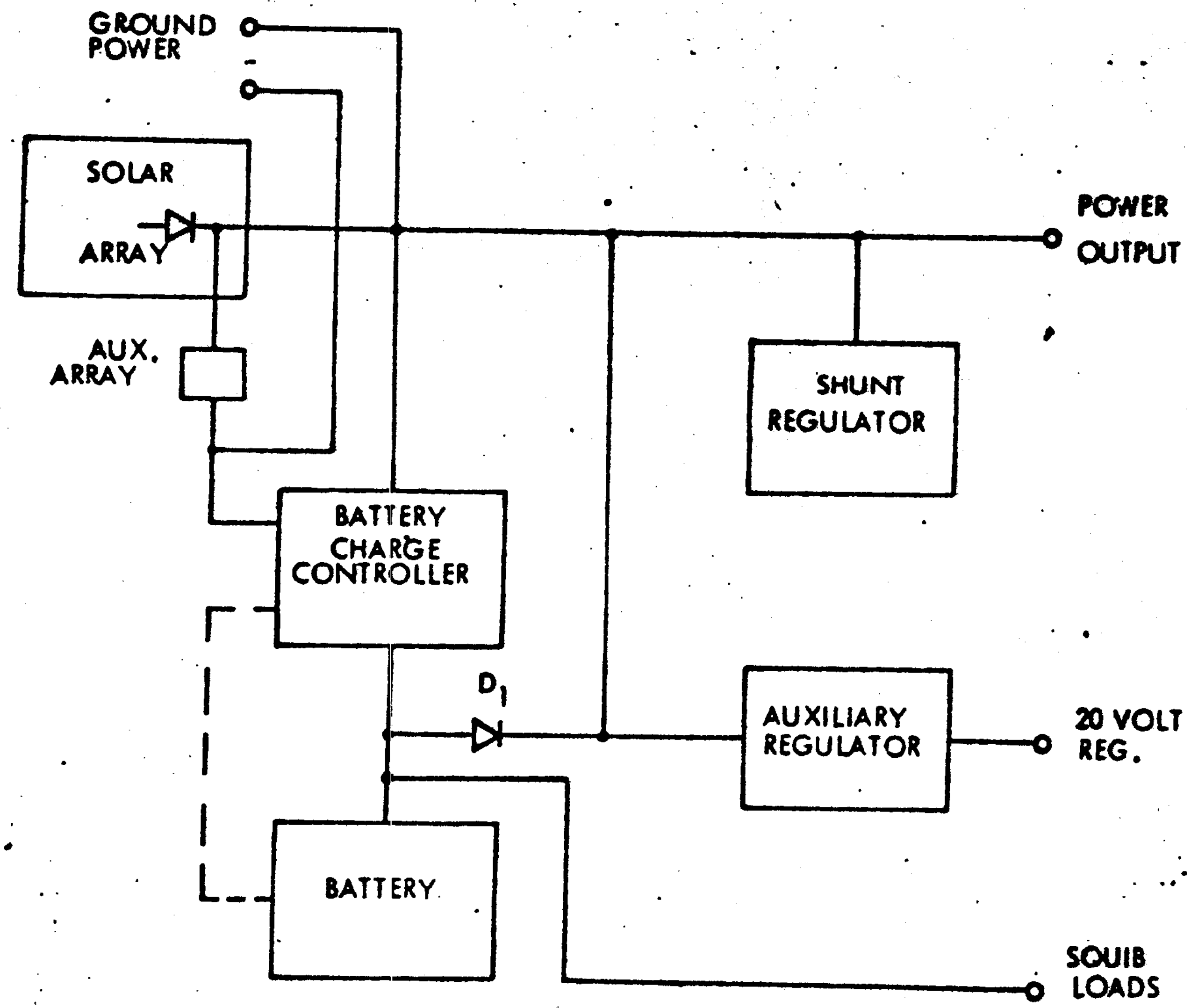


FIGURE 3.2.2-1

3.2.2.2.1 Solar Array

The Lunar Orbiter solar array consists of four identical panel structures, each having a 13.1 square foot area upon which are mounted 2714, 2cm x 2cm, N-on-P silicon solar cells. Each panel has five diode isolated circuits as follows:

(a) Three circuits consist of 104 series connected six cell modules; the cells within a module being connected in parallel.

(b) One circuit contains 104 series connected eight cell modules; the cells within a module again connected in parallel.

(c) One circuit, the auxiliary solar cell patch, consists of 10 solar cells connected in series.

The four module circuits on each panel are connected in parallel and the resultant spacecraft main array is then comprised of 104 solar cells connected both in series and parallel. The auxiliary solar cell circuit raises the potential of the solar array bus approximately 3 volts, and provides the current required to saturate the main pass transistor of the charge controller. Solar cell modules have blue reflecting filters and are 9.5% efficient at 0.46 volts and 25°C, when illuminated with 139.6 mw/cm^2 of air mass zero sunlight.

The total array weight, including the panel structures but excluding stowage hardware, i.e., actuating bracket, stowage brackets, pin pullers, etc., is 54.2 pounds. Stowage hardware weight varies from 3.40 pounds to 4.96 pounds per panel depending on the panel, the total weight is 16.32 pounds, (including actuators).

3.2.2.2.2 Storage Battery

Energy from the array is stored for night-time use in a 20 cell 12 ampere-hour, nickel-cadmium battery. The battery consists of two

3.2.2.2.2 Storage Battery (Cont.)

identical ten-cell battery modules, the envelope for one module being 7.5" x 6.2" x 6.25" high. The total battery weight is 29.96 pounds. Charge rate is determined by cell temperature and state of charges.

3.2.2.2.3 Charge Controller

The purpose of the charge controller is to protect the battery from over-current, over-voltage and over-temperature conditions while it is being re-charged. It does this by monitoring battery charge current, battery temperature and battery terminal voltage, and operating on the charging current and charging voltage. Under normal conditions the maximum charging current is limited to 2.85 ±0.15 amps; however, should the battery temperature exceed 125° ±5°F, the charging current is limited to 0.3 ± 0.1 amps. In addition to the above controls, the voltage to which the battery may be charged is limited as a function of battery temperature in the manner shown in Figure 3.2.2-2. The charge controller box also contains current telemetry circuitry comprising current-to-voltage transducers and a 1 KC inverter.

The charge controller envelope is 5.12" x 5.44" x 6.62" high and the weight is 4.09 pounds.

3.2.2.2.4 Shunt Regulator

The purpose of the shunt regulator is to limit the solar array bus voltage to a maximum of 31.0 volts. It does this by sensing the subsystem output voltage and when this tends to exceed the upper limit, the regulator turns on and by-passes a sufficient amount of array current to maintain operation at the required point on the solar array current-voltage characteristic.

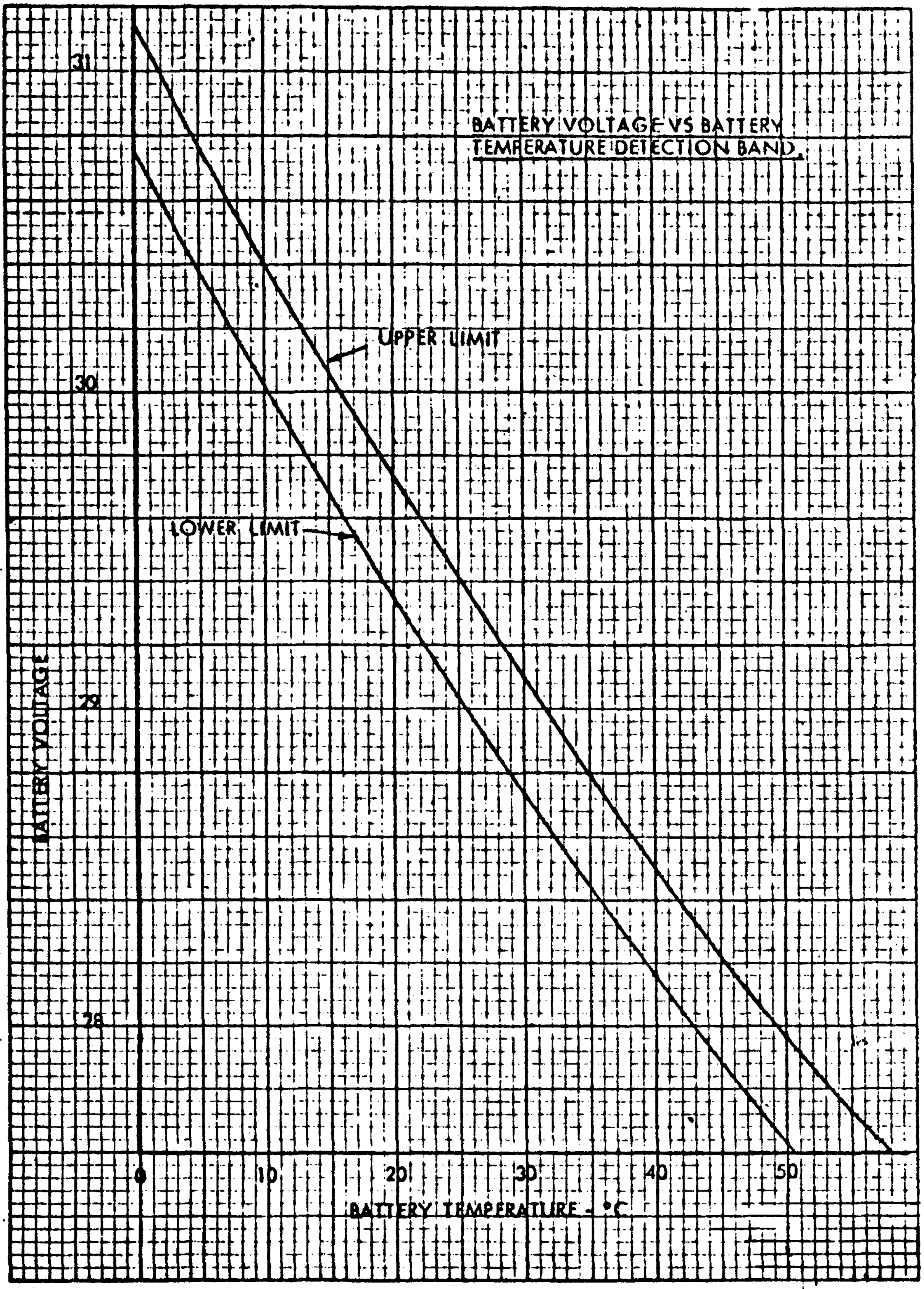

BATTERY VOLTAGE VS BATTERY TEMPERATURE DETECTION BAND
UPPER LIMIT
LOWER LIMIT
BATTERY VOLTAGE
31
30
29
28
0
10
20
30
40
50
BATTERY TEMPERATURE - °C

FIGURE 3.2.2-2

3.2.2.2.4 Shunt Regulator (Cont.)

Two auxiliary regulators are included in the shunt regulator electronic box, both provide a closely regulated 20 volt d.c. output. One regulator supplies the temperature and voltage telemetry circuitry, the other supplies the charge controller voltage and temperature sensing circuits, and the current telemetry inverter.

The shunt regulator control circuitry and the auxiliary regulators are contained in the shunt regulator electronic box which has an envelope 4.25" x 3.73" x 7.6" high. The weight of the box is 2.3 pounds. The shunt regulator power transistor assembly is installed on the outboard face of the equipment mounting deck adapter ring. The transistor cases will be looking at space in a direction radially outward in a plane parallel to the deployed solar panels. The dimensions of the assembly are 22.9" x 6.5", radius of curvature 28.82", and the weight of the assembly is 1.12 pounds.

The emitter resistor assembly is fastened to the tank deck and occupies a space approximately 14.76" long x 4.6" wide x 0.75" high. The weight of the emitter resistor assembly is 0.48 pounds. The power resistor assembly is mounted on the omni antenna boom and is approximately 20" long x 10" wide. Its weight is 1.74 pounds

3.2.2.3 Instrumentation

The power subsystem includes sensors and circuitry to provide signals for instrumentation of the parameters in Table 3.2.2-1.

3.2.2.3 Instrumentation (Cont.)

TABLE 3.2.2-1

POWER SUBSYSTEM INSTRUMENTATION

	Range	Measurement Tolerance
Panel Temperature (one per panel	$-200^{\circ}F$ to $+250^{\circ}F$	$\pm 5^{\circ}F$
Array Total Current	0 - 15 amps d.c.	$\pm 2\%$ at $25^{\circ}C$
Array Voltage (Subsystem Voltage)	0 - 40 volts d.c.	$\pm$ 1%
Battery Temperature (one per Module)	$+20^{\circ}$ to $135^{\circ}F$	$\pm 2^{\circ}F$
Battery Current	-7.5 to +7.5 amps	$\pm 2\%$ at $25^{\circ}C$
Battery Voltage	0 to 40 volts d.c.	$\pm$ 1%
D. C. Subsystem Load Current	0 - 15 amps d.c.	$\pm 2\%$ at $25^{\circ}C$
Shunt Regulator Current	0 - 15 amps d.c.	$\pm 2\%$ at $25^{\circ}C$
Auxiliary Regulator Voltage Supplies	0 - 25 volts d.c.	$\pm$ 1%

3.2.2.4 Performance

3.2.2.4.1 Output Voltage

When the array is illuminated and oriented within $\pm 5^{\circ}$ of the normal to the sun's rays, the steady state d.c. output voltage will not be less than 26.6 volts nor more than 31.0 volts. A necessary condition is that the power demand on the solar array should not exceed 185 watts during the first 0.7 hours of the daylight period of the final lunar orbit. When the spacecraft is using battery power the steady state d.c. output voltage will not be less than 22.0 volts nor more than 27.0 volts. When the subsystem is providing the power for squib loads, the output voltage will not fall below 21.5 volts. The voltage between the power output connector pins used to supply the squib loads and the spacecraft single point structure ground will not be less than 22.0 volts nor more than 27.7 volts when the subsystem is providing the power for the squib loads.

3.2.2.4.2 Output Power

The solar array output power, as a function of solar array temperature, at the end of a thirty day lunar orbit mission is given in Figure 3.2.2-3. The estimated solar array output at the end of a thirty day photographic mission, under worst case conditions, is shown in Figure 3.2.2-4. The variations in the solar array output are due to the variations in the array temperature as it orbits the moon. The array output is lowest when the array is hottest, this occurs when the spacecraft is over the subsolar point. Thus, changes in orbit will change the shape of the curve shown in Figure 3.2.2-4.

At present requirement is that the battery depth of discharge during the boost and translunar phases of the mission not exceed 60% and while in lunar orbit the depth of discharge shall not exceed 40%. Latest estimates of the spacecraft loads during launch through sun acquisition are 103.2 watts. This gives a 59% depth of discharge, assuming the maximum time from launch to sun acquisition is 1.5 hours. The present estimate of the maximum continuous spacecraft load in lunar orbit is 235.5 watts during the readout mode, with 117.7 watts during orbital night time. Assuming a maximum night time period of 0.9 hours, this gives a maximum battery depth of discharge in lunar orbit of 39.8%.

3.2.3 Communications Subsystem

3.2.3.1 Introduction

The Lunar Orbiter communication subsystem is an S-band system compatible with the existing DSIF network. The communication subsystem is designed to provide the following capabilities:

1) Receive, decode, and verify command messages.
2) Transmit spacecraft performance data, lunar environmental data, and photographic data.

3.2.3.1 Introduction (Cont.)

3) Operate in two transmitting power modes - low power mode during acquisition, tracking, ranging, and telemetry data transmission; and a high power mode during photographic data transmission periods.

4) Provide by ground command the ability to select the spacecraft transmitting power mode and to turn transmitting system on and off.

The communication subsystems are designed to provide a 6 db safety margin in received signal strength at the DSIF stations for all links.

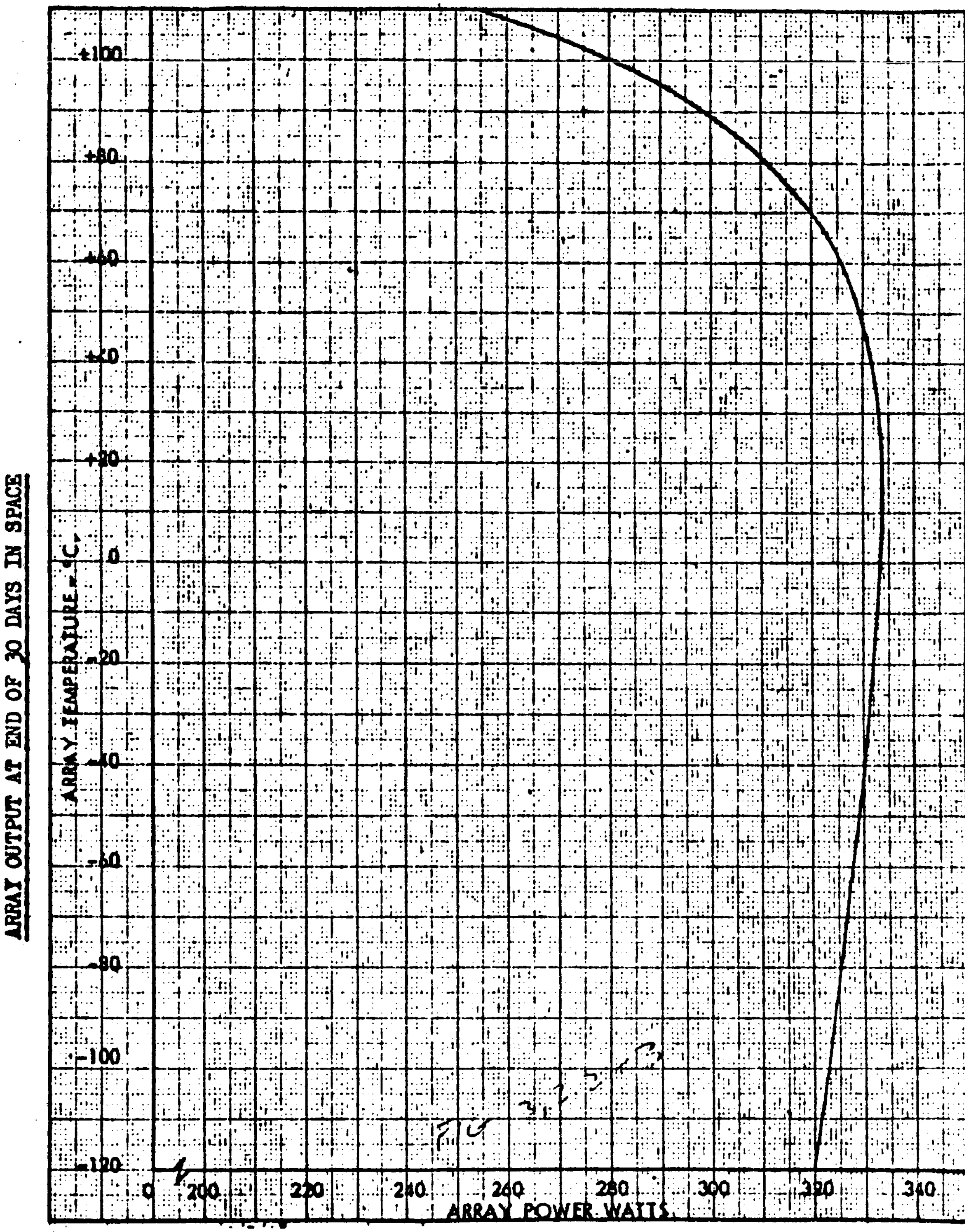
3.2.3.2 Description

The communication subsystem of the Lunar Orbiter, Figure 3.2.3-1, has been designed to transmit to the DSIF stations performance telemetry (TM), photographic pictures (video), and ranging information. In order to permit the transmission of this data, four separate communication subsystems were incorporated into Lunar Orbiter.

1) Performance telemetry subsystem
2) Photographic data subsystem
3) Ranging subsystem
4) Command subsystem

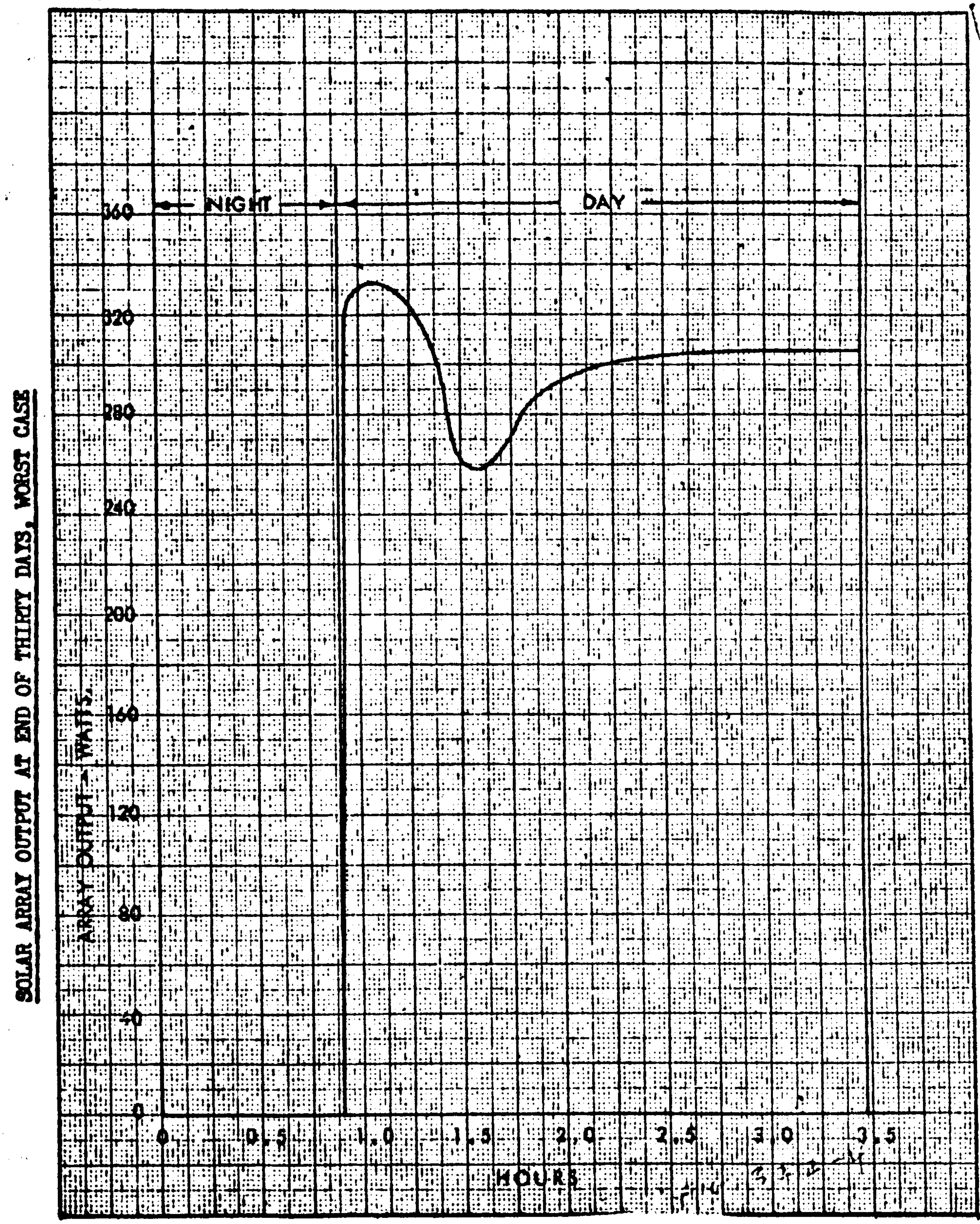
The command subsystem is an up-link system and is required in order to command and control the spacecraft from the ground. The data transmission from the spacecraft is grouped into three data or modulation modes:

Mode 1	Ranging and/or TM
Mode 2	Video and TM
Mode 3	TM only



ARRAY OUTPUT AT END OF 30 DAYS IN SPACE
ARRAY TEMPERATURE - °C
+100
+80
+60
+40
+20
0
-20
-40
-60
-80
-100
-120
0
200
220
240
260
280
300
320
340
ARRAY POWER WATTS

FIGURE 3.2.2-3



SOLAR ARRAY OUTPUT AT END OF THIRTY DAYS, WORST CASE
ARRAY OUTPUT - WATTS
NIGHT
DAY
360
320
280
240
200
160
120
80
40
0
0
0.5
1.0
1.5
2.0
2.5
3.0
3.5
HOURS

FIGURE 3.2.2-4

COMMUNICATIONS SUBSYSTEM

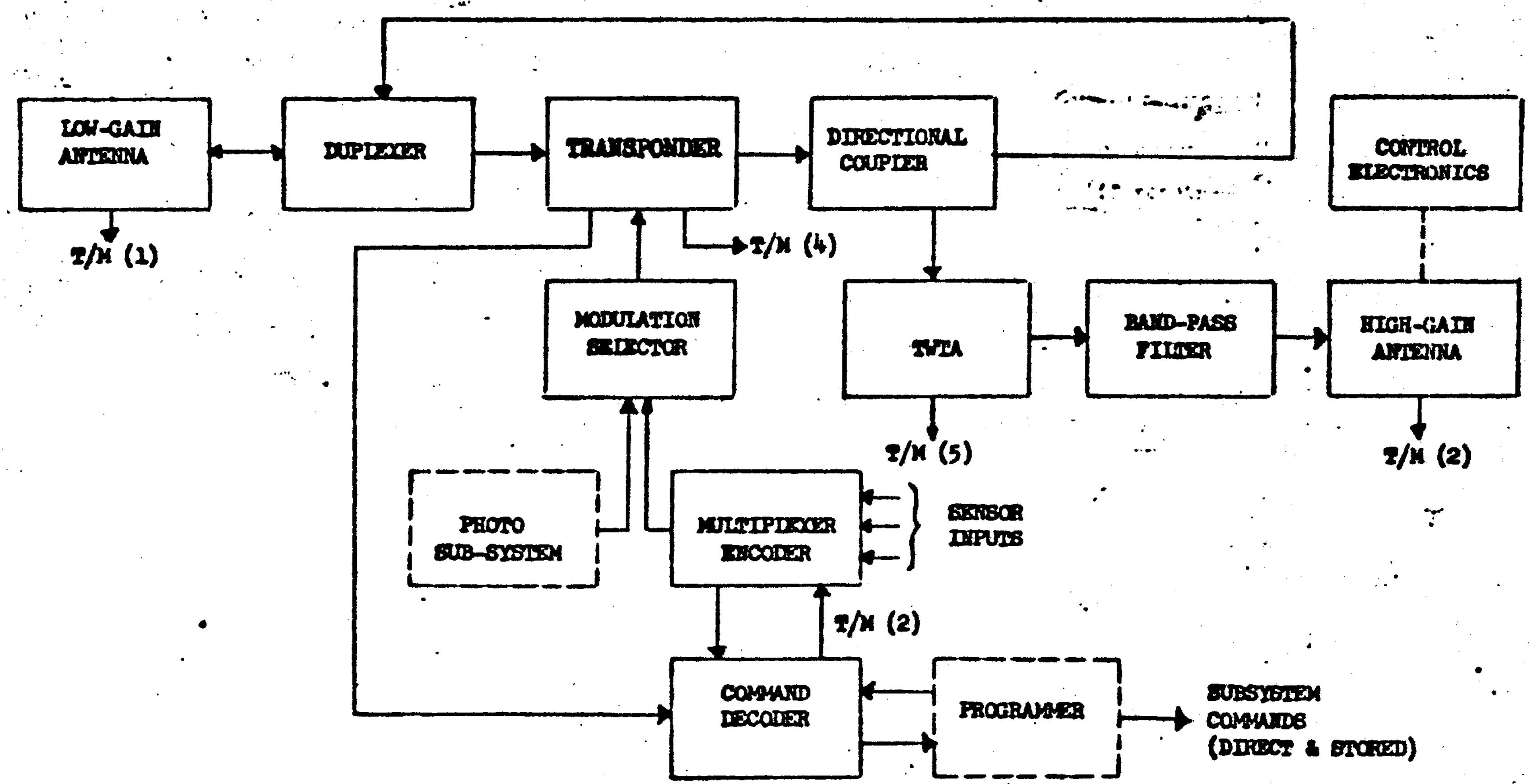


FIGURE 3.2.3-1

3.2.3.2 **Description (Cont.)**

A discrete command is required to select a particular modulation mode. For standard operating procedures the selection of a particular modulation mode automatically selects the desired transmitter power and antenna.

3.2.3.2.1 **Performance Telemetry**

Data necessary to indicate the status of the spacecraft components and subsystem operation is provided by performance telemetry. The spacecraft performance telemetry system consists of a single 1152 bit digital frame which is transmitted to earth at 50 bits per second (bps). The 50 bps data is differentially coded (NRZ) and diphase (PSK) modulated upon a 30 KC subcarrier. This subcarrier in turn is phase modulated upon the 2295 MC S-band carrier.

The analog word length is 9 bits; 8 bits are used for the analog data with the ninth bit being the compliment of the eighth bit. This provides a change in level at least every 9 bits and will assure bit synchronization in the receiving ground equipment. The 1152 bit frame consists of 128 separate 9 bit words.

Analog information from the transducers is sampled by the analog multiplexer at the appropriate time and is presented to the analog-to-digital (A/D) coverter for digitalization (coded in binary form). This digital signal is then inserted into the 1152 bit telemetry frame. Digital information derived directly from digital sources (not requiring storage) is sampled by the digital multiplexer and directly inserted into the frame. Frame synchronization is obtained by correlating a 43 bit legendre code (inserted within the frame) while bit and phase synchronization are extracted from the 30 kcps signal.

3.2.3.2.1 Performance Telemetry (Cont.)

The telemetry system has the capability of transmitting continuously, either separately or simultaneously with the ranging or video signals.

3.2.3.2.2 Photographic Data Transmission (Video)

The modulation method for the photographic data transmission uses a vestigial side-band AM subcarrier which in turn phase modulates the carrier. This provides d.c. response and maintains the maximum phase deviation of the carrier below 4 radians. This is accomplished as follows:

The video signal from the scanner is mixed with synchronization and blanking pulses and then passes through a dc to 230 kc filter. The dc to 230 kc band of signals is then translated, using a balanced (double sideband suppressed carrier) modulator and a 310 kc carrier to the 80 kc to 390 kc band. The resultant double sideband signal is then passed through a vestigial sideband filter that has 6 db attenuation at 310 kc. This band of frequencies is then mixed with the 30 kc telemetry subcarrier and a coherently derived 38.75 kc pilot carrier before phase modulating the 2295 mc carrier. At the receiver the 50 mc I.F. signal is taken through a frequency compression feedback demodulator (FMFB) and a 0 to 390 kc low pass filter. Three band pass filters then separate out the 30 kc telemetry subcarrier, and the 38.75 kc carrier is multiplied by 8 resulting in a coherent 310 kc signal which is used to translate the 80 kc to 390 kc band back down to the original 0 to 230 kc band.

The wide band width (3-1/3mc) of the transmitted video signal requires the use of the high power (10 watt) traveling wave tube amplifier (TWTA) and high gain(23.5 db) antenna.

3.2.3.2.3 Ranging Subsystem

The range to the spacecraft is measured by using the DSIF pseudo-random (PN) range code system. The system is intended primarily for operation while in lunar orbit but may be used during the translunar trajectory period.

The pseudo-random code bits are synchronously detected at the output of the wide band IF amplifier in the spacecraft transponder and remodulated onto the down link carrier for transmission to earth. At the DSIF site the code bits are again synchronously detected in the 10 mc wide band detector. The output from this detector is then fed to the range code correlator where the total transmission time is determined to an estimated accuracy of ± 0.1 microseconds (± 15 meters). A discrete command is required to turn on and to turn off the ranging system.

3.2.3.2.4 Command Subsystem

The spacecraft receiver is required for command reception. The command system receives and stores in a command decoder a 26 bit digital word, retransmits the stored word via the telemetry system for ground verification, and provides a command signals to the various subsystems. The command decoder is a modified unit from the Relay satellite and consists of two parallel redundant decoders operated by various combinations of multitone signals. Four tones (discrete subcarrier frequencies) plus the FSK command signal are required to operate the system. Two of these tones, designated the enable tones, are required to select the desired decoder while the other two tones, called the execute tones, are used to transfer the commands (once they have been verified) from the decoder into the programmer. The simultaneous occurrence of both enable tones selects the programmer redundancy mode while the absence of the tones clears the decoders.

3.2.3.3. Subsystem Specifications

Low Gain Antenna

Gain	0 db nominal in XZ plane
Beamwidth	-6 db, $\pm 60^{\circ}$, from XZ plane (Coverage in excess of 95% of radiation sphere)
Polarization	Right hand circular with axial ratio of 1.59 or less within the 3 db beamwidth
Mounting	Deployable boom upon command
Operating frequency	2295 $\pm$ 5 mc, transmit 2115 $\pm$ 5 mc, receive
Impedance	50 ohm nominal
VSWR	1.6 to 1 during transmit mode 3.0 to 1 during receive mode
Type	Biconical horn
Boom length	82 inches long

High Gain Antenna

Gain	+23.5 db minimum at spacecraft interface
Beamwidth	10 degrees solid angle minimum, centered to $\pm$ 1 degree of mechanical boresight of the antenna as measured at spacecraft interface
Type	Circular parabolic dish illuminated by crossed dipole feed
Sidelobe suppression	At least 15 db below main beam maximum
Polarization	Right hand circular with axial ratio of 1.59 or less within the 10 degree beamwidth
Operating frequency	2295 $\pm$ 5 mc
Impedance	50 ohms nominal
VSWR	1.3 to 1 over operating frequency band

3.2.3.3.0 **Subsystem Specifications (Cont.)**

Steering	Can be steered about boom axis in 1 degree increments by ground command
Mounting	Deployable boom steerable 360 degrees about the boom axis capable of being preset $\pm 5^{0}$ about both roll and yaw axis.
Pointing loss	0.2 gyro limit cycle (attitude control)

Transponder

Receiver sensitivity	149 dbm
Receiver frequency band	2115 ± 5 mc
Transmitter frequency	2295 mc
Output power	+ 26 dbm (400 mw)
Tracking loop noise bandwidth	$2B_{LO}$ = 100 cps
Duplexer	
a. Type - Coaxial	0.5 db
b. Insertion loss	0.5 db
Directional coupler	
a. Type - coaxial	
b. Insertion loss	0.5 db
c. Coupling Figure	16 ± 1 db (10 mw to TWTA)
d. Directivity	25 db

Modulation Modes

Carrier and Range Tracking (2-way) links....................PM

Performance telemetry link...............PSK(diphase)/PM

Photographic data (video) link......Vestigial sideband (AM)/PM

Command data link....................................FM/PM

3.2.4 **Attitude Control Subsystem**

3.2.4.1 **Functions**

The functions of the Attitude Control Subsystem are to:

3.2.4.1 Functions (Cont.)

a. Acquire and maintain an inertially fixed reference attitude during the coast and picture readout phases of the mission as required by thermal control, solar power and the high gain antenna. The inertial reference during the 30-day Photographic Mission shall be established by a line to the sun and a line to Canopus. The inertial reference during the subsequent Extended Mission shall be established by a line to the sun only.

b. Perform attitude maneuvers to acquire and maintain attitude required for velocity control adjustments for midcourse correction, orbit injection and orbit transfer. Re-acquire initial reference attitude subsequent to velocity adjustment.

c. Perform attitude maneuvers to acquire and maintain attitude required for photography, including attitude adjustment for image motion compensation. Re-acquire initial reference attitude subsequent to photography.

d. Compute spacecraft velocity changes from the measurement of acceleration during engine firing.

e. Initiate all spacecraft events either by stored program commands or earth generated commands received via the communication subsystem.

f. Provide spacecraft time to the photo subsystem, and timing reference for the communication subsystem.

g. Provide telemetry data to identify spacecraft attitude, velocity, time, and stored program status.

3.2.4.2 Description

The major elements of the ACS (attitude control SVB system) are the flight Electronics control assembly, the Inertial Reference Unit, Sun

3.2.4.2 Description (Cont.)

Sensors, Star Tracker, Switching Assembly, Thrust Vector Control, Reaction Control which includes N_2 gas supply and plumbing and thrusters. Figure 3.2.4-1 is a block diagram which shows the over-all arrangement and signal flow.

3.2.4.2.1 Flight Electronics Control Assembly

The Flight Electronics Control Assembly contains the programmer and closed loop electronics.

3.2.4.2.1.1 Programmer

The Programmer is a low speed digital data processing machine with 21 bits, serially read and with random access to the memory. It has 128 words. It controls 86 plus discrete functions. It has redundant clocks providing timing in 0.1 second increments up to 29.1 hours. Clock stability is .0001% for 8 hours. It can provide at least 16 hours of control over a photographic mission from stored program commands. Maneuvers up to 360° (in increments as small as 0.011°) and velocity changes up to 3000 feet per second (in increments as small as 0.1 FPS) are provided by 15 bit registers. The programming flow diagram is illustrated by Figure 3.2.4-2. It has been designed to have programming versatility to handle mission changes as well as subsystem control changes. It has been organized to operate normally in the stored program mode to accomplish the primary photo mission objectives and to utilize real time command mode as a mission back-up and for auxiliary functions.

The Programmer will proceed with the execution of a stored program, bringing commands sequentially from memory, completing them and continuing with a "compare time to next event". The stored program must be periodically updated by ground control to maintain mission

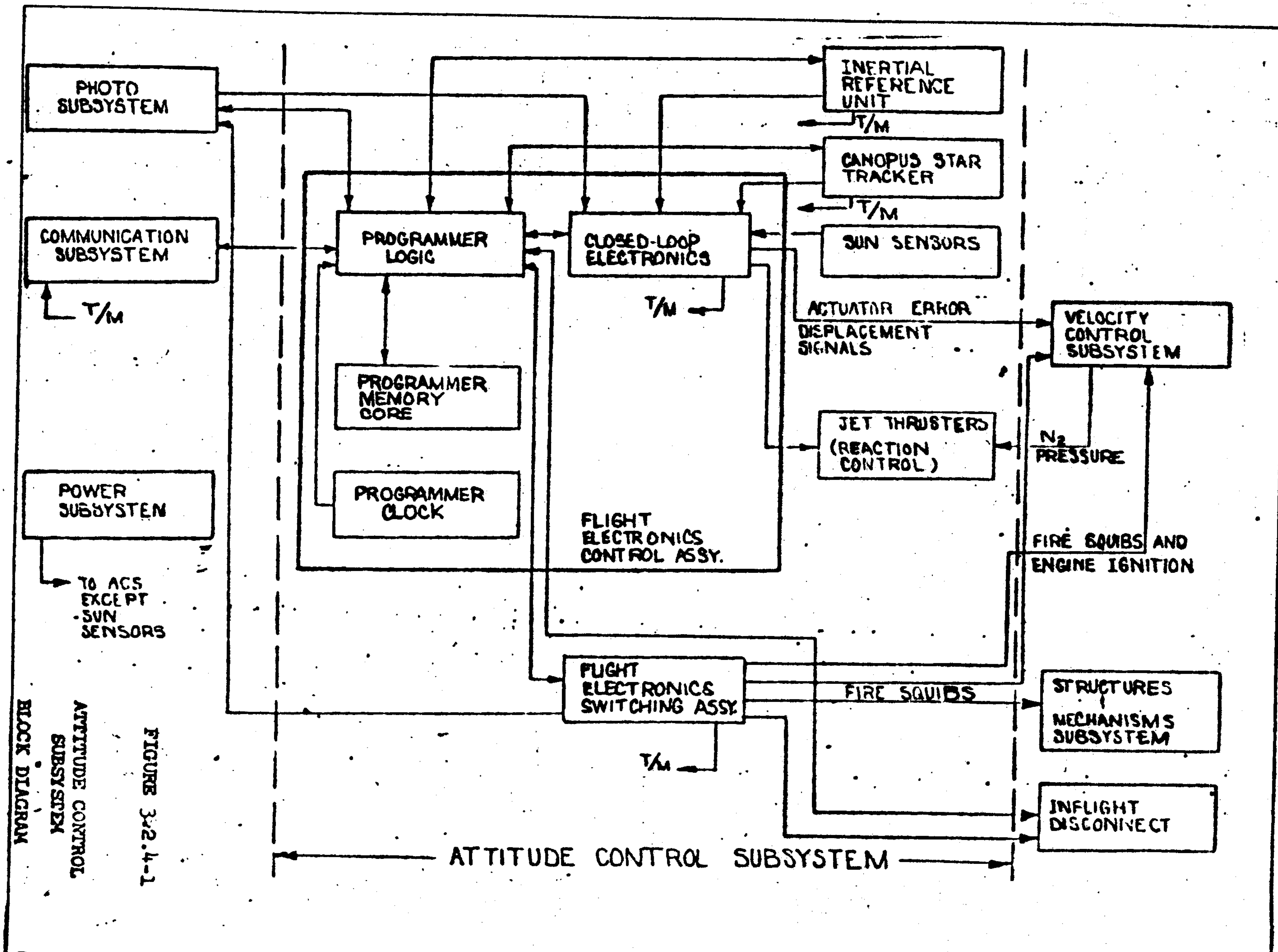


FIGURE 3.2.4-1

ATTITUDE CONTROL SUBSYSTEM

BLOCK DIAGRAM

LUNAR ORBITER PROGRAMMING FLOW DIAGRAM

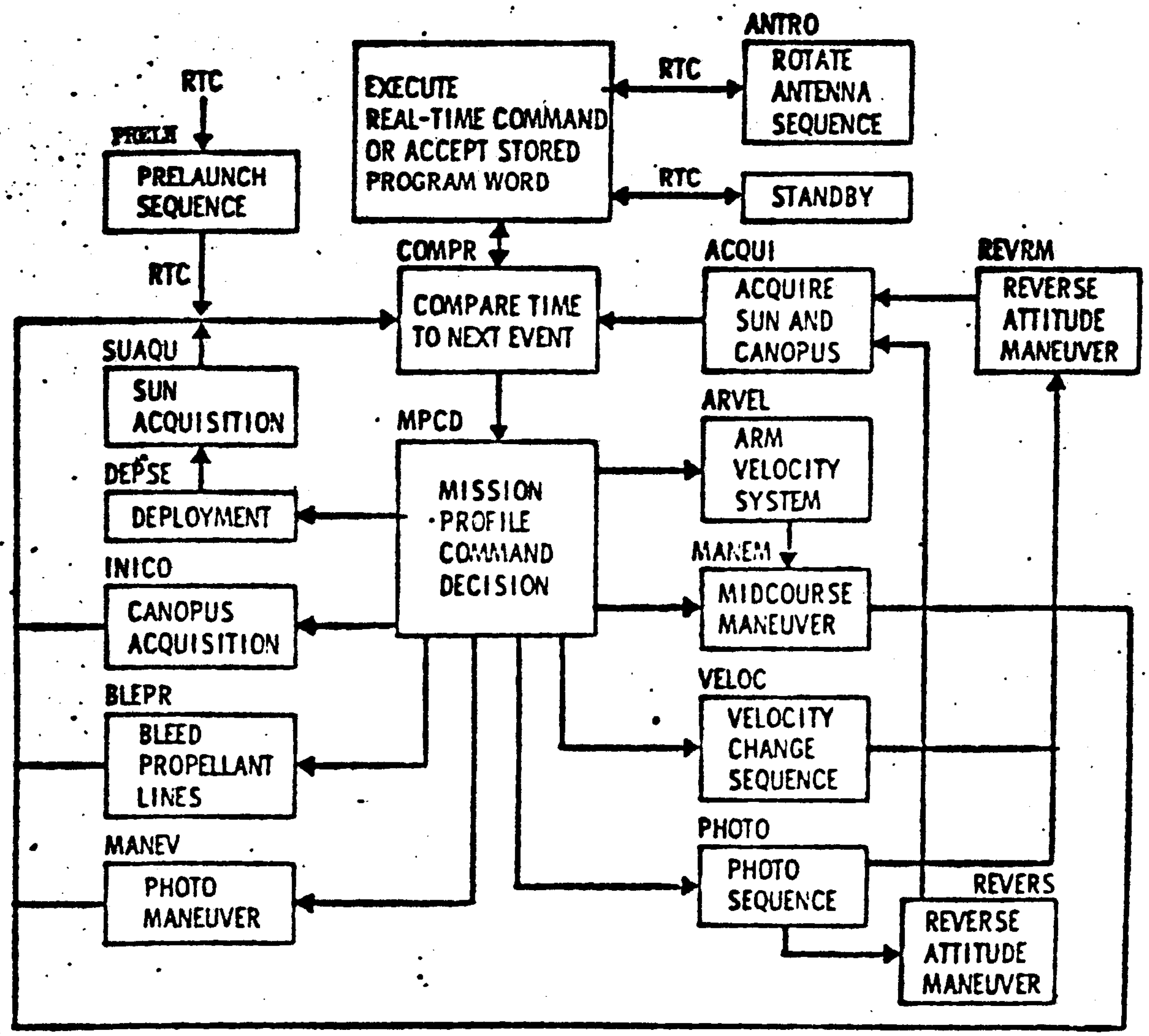


FIGURE 3.2.4-2

3.2.4.2.1.1 Programmer (Cont.)

continuity and validity.

3.2.4.2.1.2 Closed Loop Electronics

The Closed Loop Electronics is contained within the Flight Electronics Control Assembly. It contains the reaction jet valve drivers, signal summing amplifiers and limiters, sun sensor amplifiers and limiters, thrust vector control signal amplifiers and compensation networks, reaction control compensation networks, slew signal generators, switching functions, and signal conditioning for certain telemetry channels.

3.2.4.2.2 Sun Sensor

The Sun Sensors provide a celestial reference for the pitch and yaw axes by providing an output signal to the Closed Loop Electronics indicative of the angular deviation from the line to the sun. Remote aft looking coarse eyes complete the spherical field of view coverage as illustrated in Figure 3.2.4-3.

3.2.4.2.3 Star Tracker

The Star Tracker provides a celestial reference for the roll axis by locking onto and tracking the star Canopus. It provides an output signal proportional to the angular deviation in roll of the line of sight to the star when it is in the tracker field of view. It also generates a recognition signal and a star intensity signal for transmission by telemetry to the ground for use in construction of a star map. A schematic of the tracker is shown in Figure 3.2.4-4.

3.2.4.2.4 Inertial Reference Unit (IRU)

The IRU senses attitude rates about the three control axes by means of three "strapped down", floated, rate integrating gyros. The operating mode for each gyro is selectable independently and may be either rate

SUN SENSOR FIELDS OF VIEW

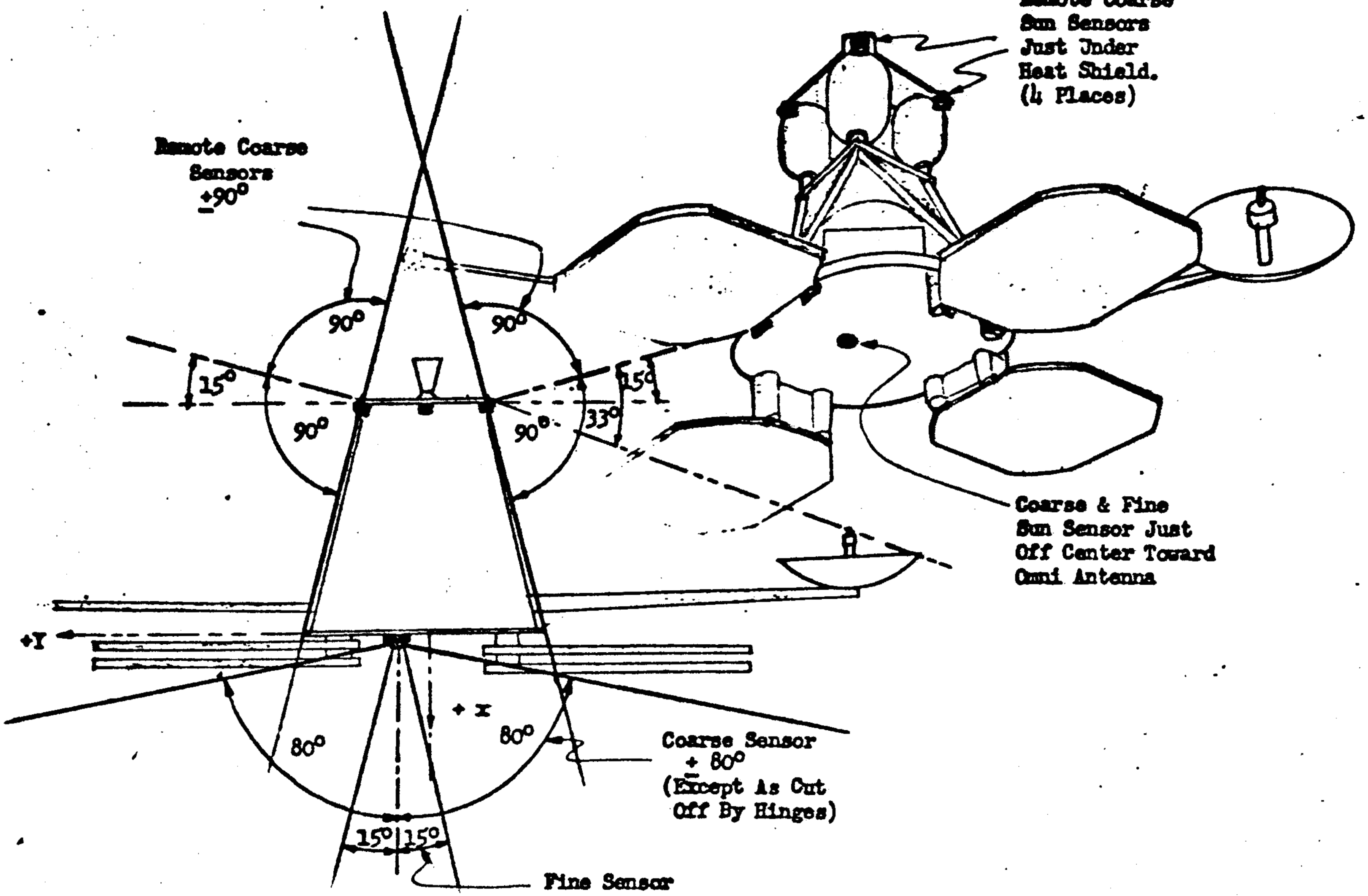
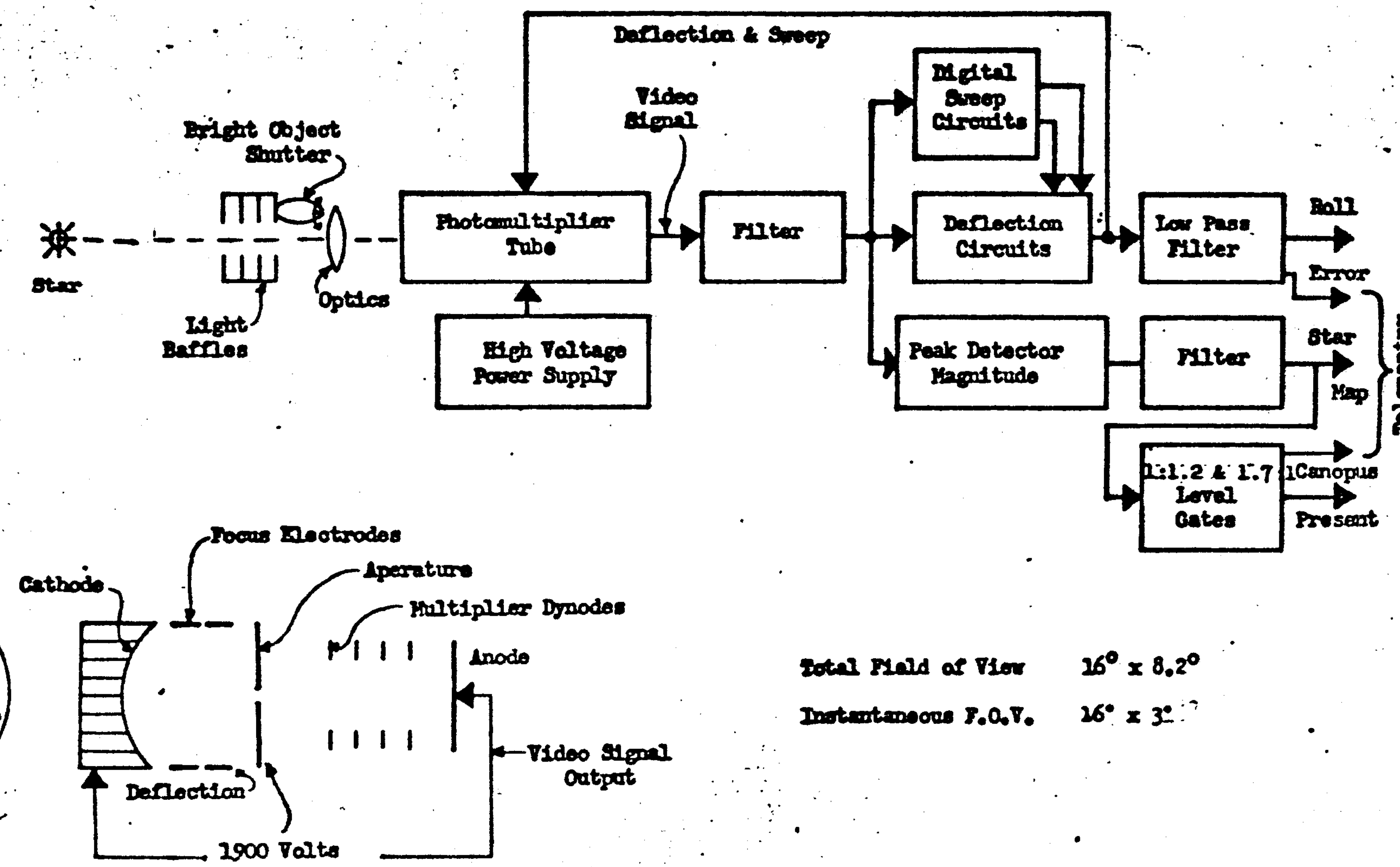


FIGURE 3.2.4-3

STAR TRACKER SCHEMATIC



Total Field of View $16^\circ \times 8.2^\circ$

Instantaneous F.O.V. $16^\circ \times 3^\circ$

FIGURE 3.2.4-4

3.2.4.2.4 Inertial Reference Unit (IRU) (Cont.)

mode or rate integrating mode. The IRU also provides for the measurement of velocity change by integrating the output of a linear accelerometer. It also provides telemetry information for transmission to the ground. A schematic of the IRU is shown in Figure 3.2.4-5.

3.2.4.2.5 Reaction Control

The control torques in roll, pitch, and yaw are provided by reaction control thrusters. A schematic diagram of the thruster arrangement and pertinent control data is contained in Figure 3.2.4-6. A plumbing system provides the distribution of pressure regulated nitrogen gas to the thrusters. The N_2 gas is stored in a supply tank and is used for the Attitude Control Subsystem and the Velocity Control system. A breakdown of the N_2 gas supply budget is contained in Table 3.2.4-1.

3.2.4.2.6 Thrust Vector Control

Control of pitch and yaw is provided by gimballing the engine mounting ring with electronic servo actuators during engine firing. The actuators are energized simultaneously with the engine control valves. Actuator position is provided at that time for telemetry.

3.2.4.3 Operating Modes

Several modes of control are provided in order to accomplish the various phases of the mission.

Celestial Hold modes uses the outputs of the sun sensor and star tracker to maintain a reference orientation. The rate integrating gyros operating in the rate mode provide damping. Each sensor is independently selectable. This is a limit cycle operation of the reaction control system.

INERTIAL REFERENCE UNIT

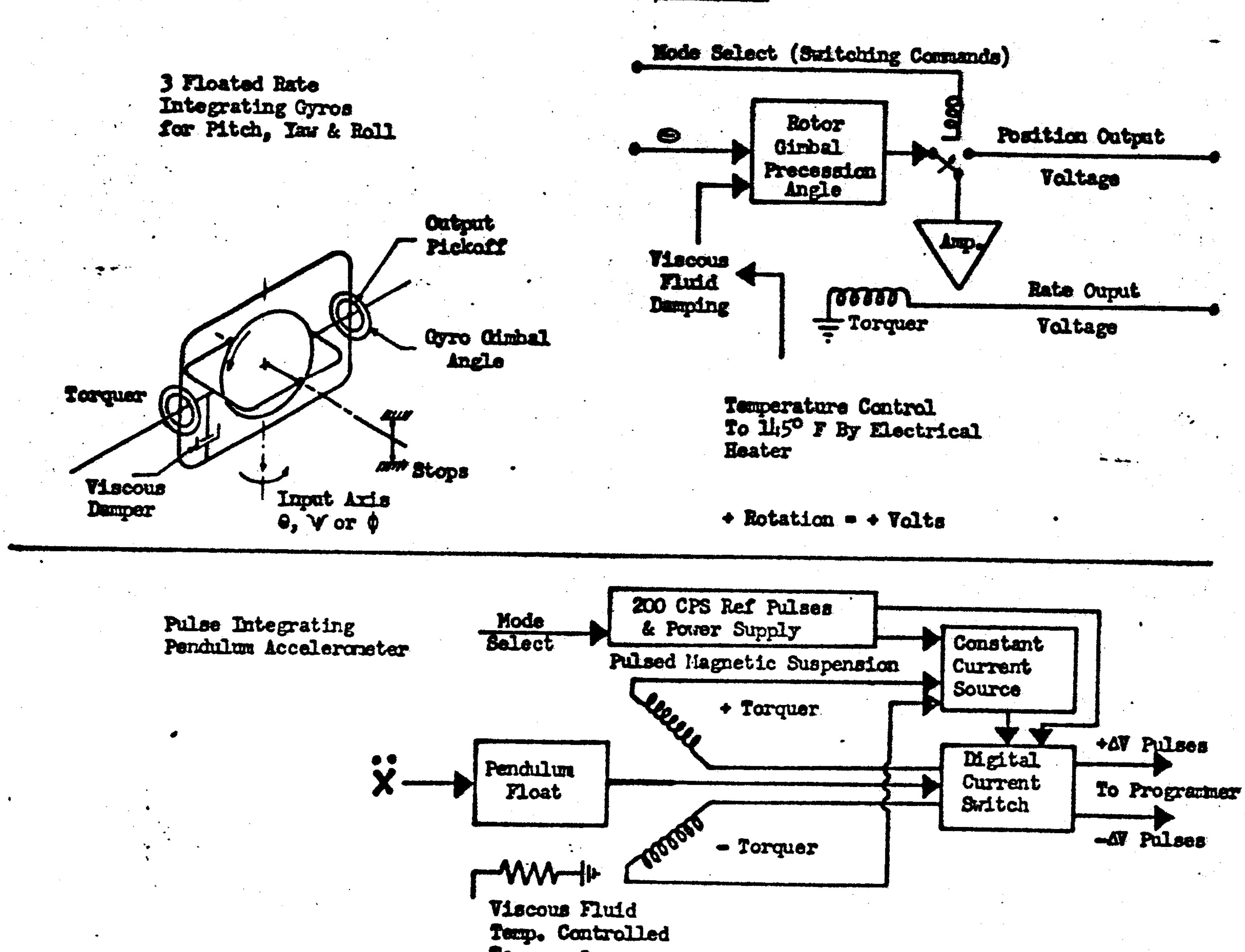


FIGURE 3.2.4-5

REACTION CONTROL DIAGRAM

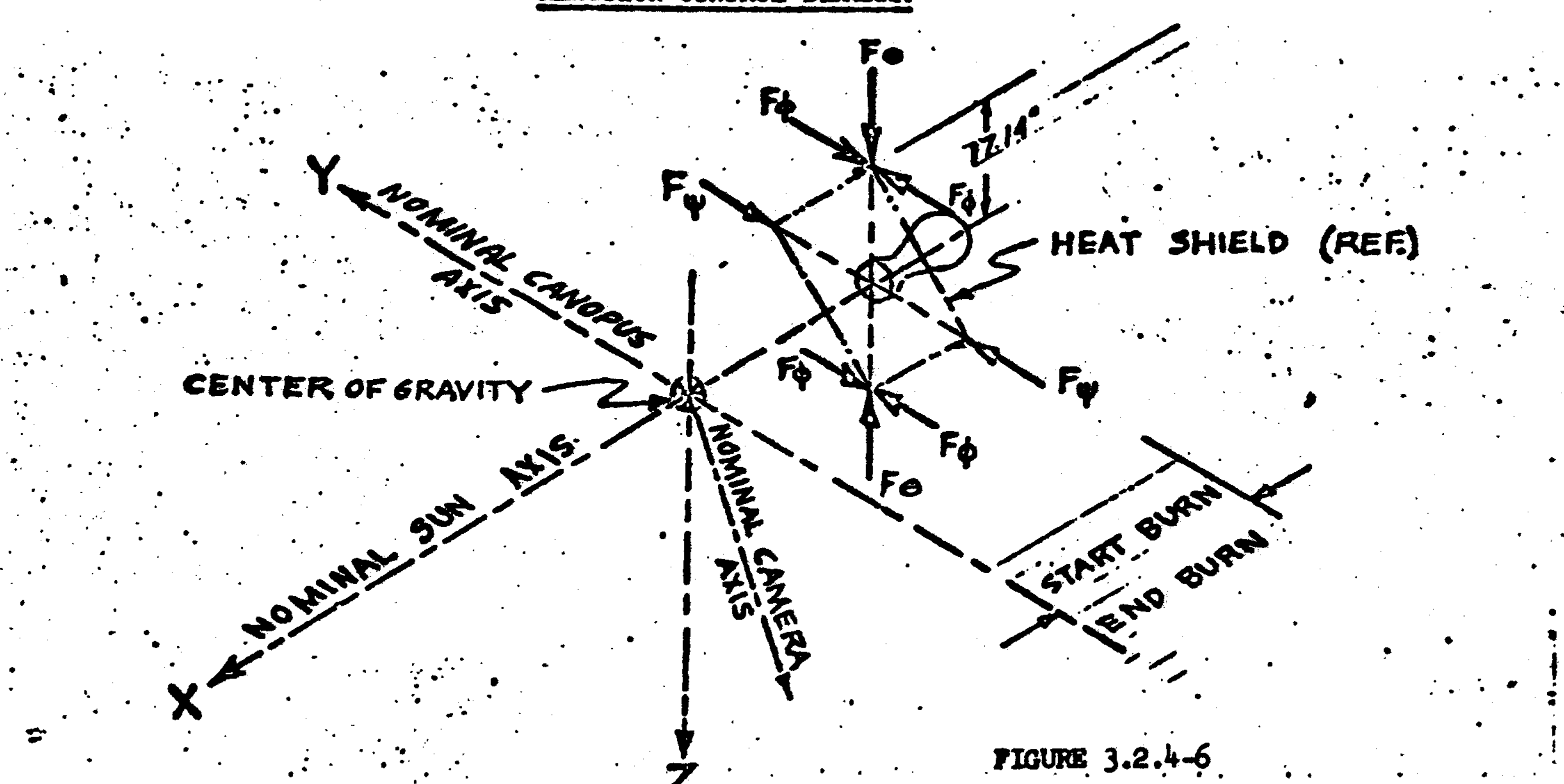


FIGURE 3.2.4-6

		INERTIA	REACTION CONTROL			THRUST VECTOR CONTROL	
		SLG-FT^2	Σ JET FORCE	LEVER ARM	ACCELERATION °/sec^2	C.G. TO GIMBAL	ACCELERATION
START BURN	PITCH	122.60	.05	31.32"	.06105	31.16"	2.12 °/s^2/°
	ROLL	83.94	.056	17.14"	.0546		
	YAW	134.31	.05	31.37"	.0557	31.16"	1.933 °/s^2/°
END BURN	PITCH	75.98	.05	41.99"	.1321	41.83"	4.59 °/s^2/°
	ROLL	74.27	.056	17.14"	.06175		
	YAW	84.50	.05	42.04"	.119	41.83"	4.73 °/s^2/°

TABLE 3.2.4-1

VELOCITY AND REACTION CONTROL NITROGEN GAS REQUIREMENTS

MISSION PHASE	NITROGEN WEIGHT ITEM	NITROGEN WEIGHT TOTAL
Photographic Mission		
(1) Initial Mission	.22	
(2) Translunar Coast	.06	
(3) Midcourse Maneuvers		
(a) 1st Midcourse	.14	
(b) 2nd Midcourse	.18	
(4) Initial Orbit Injection	.14	
(5) Final Orbit Transfer	.09	
(6) Photo Maneuvers (12)	1.19	
(7) Photo Transmission (10 days)	.68	
(8) Lunar Orbit Coast (17 days)	.12	
(9) Celestial Reacquisition	.84	
(10) Disturbances	.14	
(11) R/C Cross Coupling	.20	
Photo Mission Total		4.00
Extended Mission		
(1) Lunar Orbit Coast	1.79	
(2) Celestial Reacquisition	.56	
(3) Disturbances	1.48	
(4) R/C Cross Coupling	.19	
Extended Mission Total		4.02
Reserve		1.98
Total Nitrogen Budget, Reaction System		10.00
Velocity Control Subsystem		
(1) Propellant Expulsion	3.84	
(2) Reserve	.16	
Velocity Control Total		4.00
Combined Systems		
(1) Leakage	.49	
(2) Residual	.09	
Total Nitrogen for Combined Reaction and Velocity Control		14.58

3.2.4.3 Operating Modes (Cont.)

Inertial Hold modes uses the rate integrating mode of the gyro to obtain position information. A lead-lag network is used to develop rate information for stabilization of the vehicle using reaction control thrusters and is a limit cycle operation of the system. Additional lead-lag networks are used for stabilization during powered flight when pitch and yaw signals are applied to the thrust vector control (TVC) actuators.

Constant Rate mode uses the integrating gyro in its rate mode to obtain rate information for control through the reaction control thrusters. When a maneuver is desired, a slew signal is summed with the gyro rate signal and reaction control rotates the spacecraft one axis at a time at a constant rate. The rate signal can be integrated by the programmer to obtain a specified change in attitude. The rate mode may also be selected without a slew command and a spacecraft rates controlled to less than an amount determined by the deadband selection. The options available are illustrated in Figure 3.2.4-7 and 3.2.4-8.

Velocity Correction mode is used to perform velocity corrections and changes required by the mission for midcourse, injection, and correction maneuvers. The attitude control system is operated in the Inertial Hold mode, as above, Roll control is obtained by the roll thrusters of the reaction control system.

3.2.4.4 Mission Phases

In order to illustrate the operational capability of the attitude control system, a typical mission sequence is described as follows.

SWITCH LINE OPTIONS

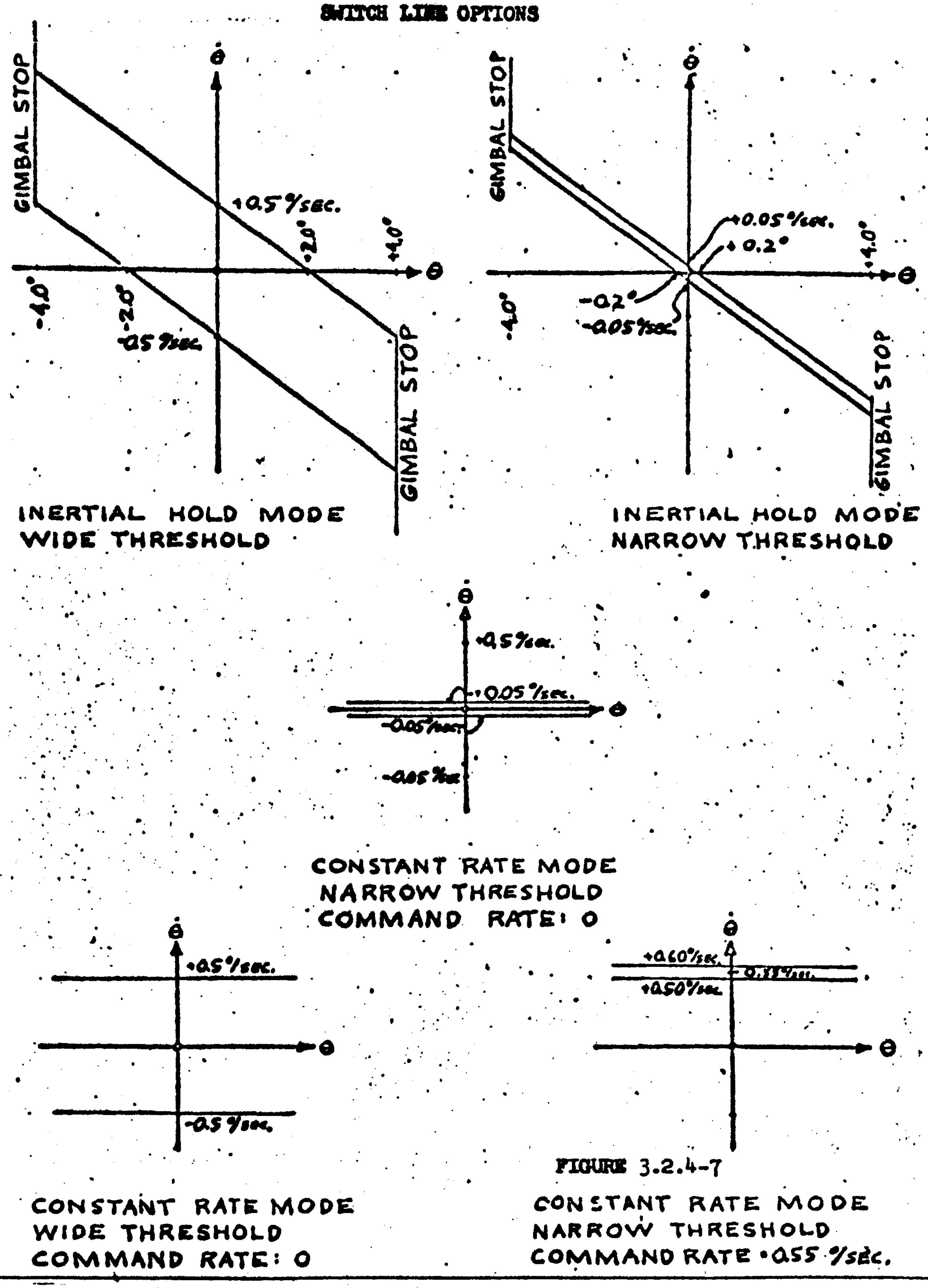
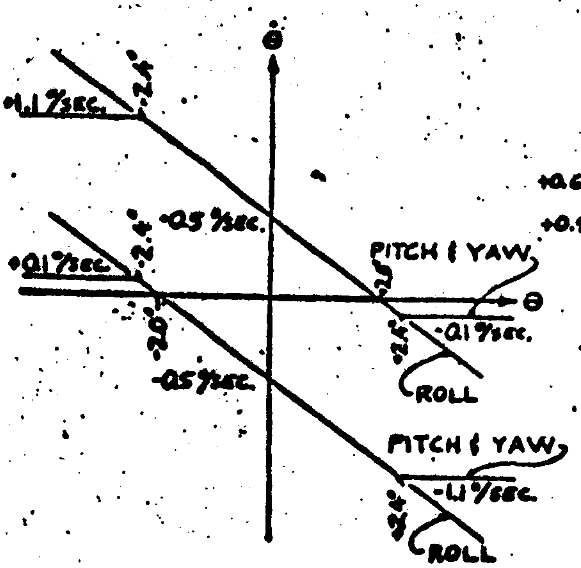


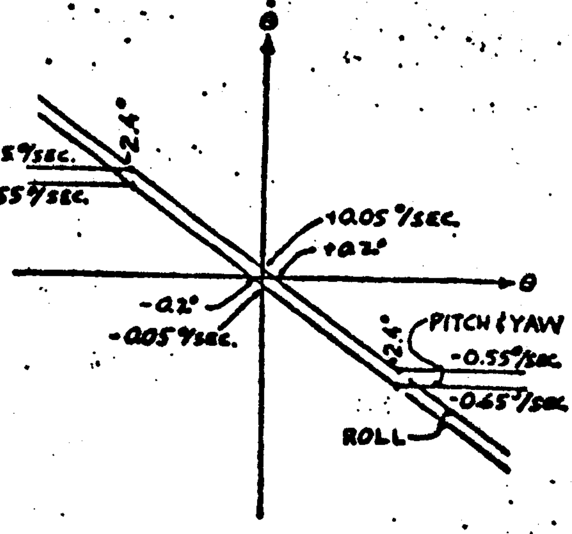
FIGURE 3.2.4-7

SWITCH LINE OPTIONS - LIMIT CYCLE MODE

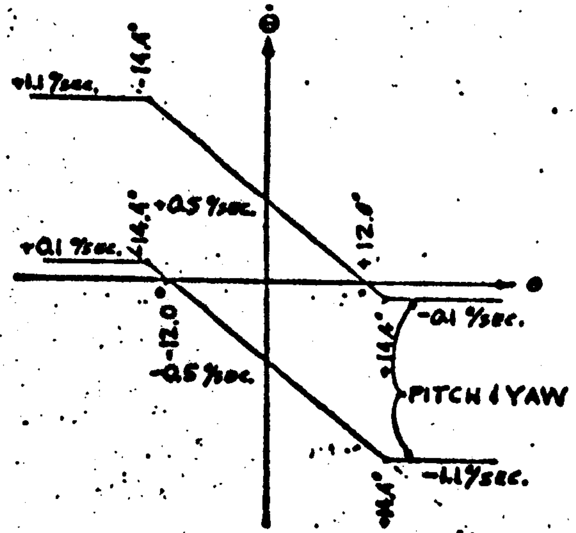
NOTE: LIMIT CYCLE RATES SHALL BE MAINTAINED BELOW 0.01 DEGREES/SECOND



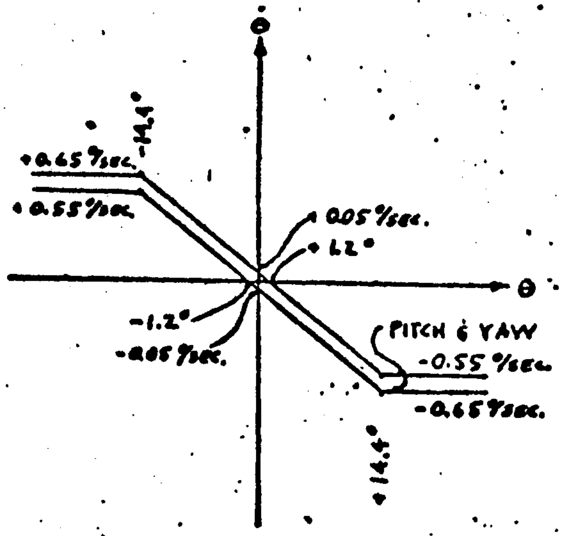
LIMIT CYCLE MODE
WIDE THRESHOLD
FINE SUN SENSOR (PITCH & YAW) —
CANOPUS SENSOR (ROLL) ——



LIMIT CYCLE MODE
NARROW THRESHOLD
FINE SUN SENSOR
CANOPUS SENSOR



LIMIT CYCLE MODE
WIDE THRESHOLD
COARSE SUN SENSOR



LIMIT CYCLE MODE
NARROW THRESHOLD
COARSE SUN SENSOR

FIGURE 3.2.4-8

3.2.4.4.1 <u>Launch</u>

The attitude control system is electrically activated from pre-launch. The gyros are held in the rate mode by an occultation inhibit command from the programmer, the deadband selection is $\pm$ 2° and the squib valves seal off the nitrogen supply from the reaction control system.

3.2.4.4.2 <u>Initial Sun Acquisition</u>

The occultation inhibit signal remains until removed by a real time command from the ground. Sun acquisition follows a sequence of commands from the programmer which deploy the solar panels and antennas, and fires the nitrogen squib valves which activates the reaction system. The programmer selects the narrow ($\pm$.2°) deadband along with the coarse sun sensors which provide total spherical coverage, and the sun is automatically acquired within 20 minutes. No roll position reference is used until after the Star Tracker is to be energized, 6 or more hours after injection which avoids operation in the Van Allen belts. Roll rate is constrained to less than .05°/sec, if the narrow deadband is selected.

3.2.4.4.3 <u>Canopus Acquisition</u>

This phase of the mission requires continuous monitoring by the ground in order to ascertain the acquisition of the star Canopus. The star tracker is turned on by the programmer. A 360° roll maneuver is commanded by the ground for the purpose of obtaining a star map via the telemetry link. The sequence of events is shown in Figure 3.2.4-9. The random roll attitude is retained by the inertial hold mode until the groun personnel identify the star and the proper magnitude meneuver to bring the star within the tracker field of view. The sequence is completed by a real time command from the ground. The spacecraft continues to operate in the limit cycle

CANOPUS ACQUISITION

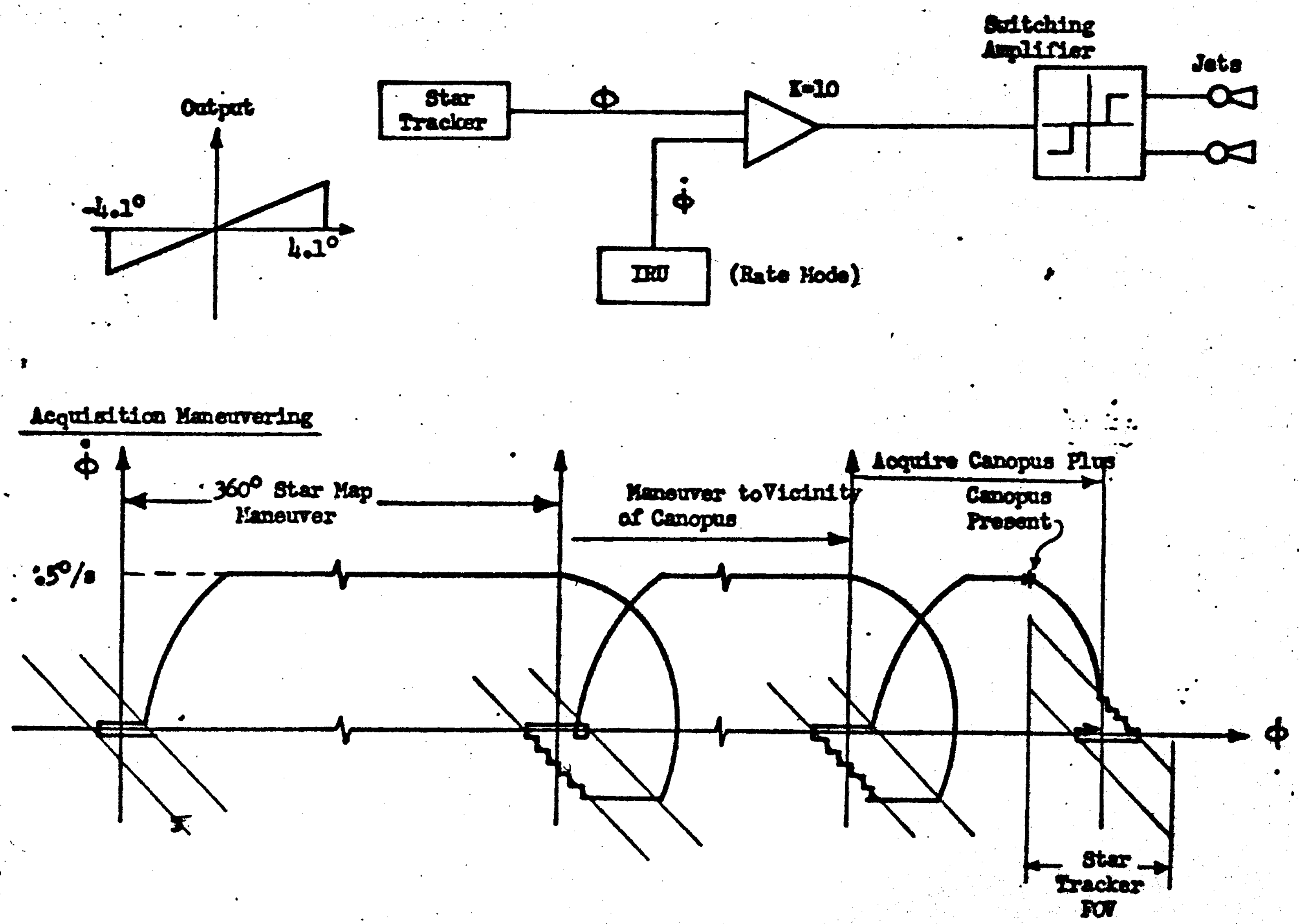


FIGURE 3.2.4-9

3.2.4.4.3 Canopus Acquisition (Cont.)

mode until further instructions are received from the ground.

3.2.4.4.4 Velocity Change Attitude Maneuver

The maneuver sequence and velocity change magnitudes desired are transmitted from the ground, verified, and stored in the programmer for execution at some later time. The velocity change engine is prepared for activation by a sequence from the programmer, which includes bleeding the lines and finally blowing the propellant squibs. The fine sun sensor is switched on and the coarse sun sensors off in preparation for precise attitude maneuvers.

The attitude maneuver is initiated from the narrow (.2°) deadband in all axes. Usually the first maneuver is a roll about the sun line. A roll slew command is switched to the reaction control system, the roll gyro is switched to the rate mode, the output is integrated by counting the pulses from the voltage-to-frequency converter. Upon receipt of a "command done" from the programmer when the maneuver magnitude equals the number stored in the maneuver register, the gyro is switched to Inertial Hold and the slew signal removed.

A wait time of 51.2 seconds is provided to allow settling to the limit cycle before initiating the second maneuver of the sequence, which could be a pitch or yaw. For a pitch maneuver, the yaw gyro is first switched to inertial hold mode, the pitch gyro to the rate mode, a slew signal is switched to the reaction control system and the spacecraft pitches at constant rate, .5°/sec, until the programmer issues a "command done" at which time the gyro is switched to the inertial hold mode. If a subsequent yaw maneuver is required, it follows in a similar manner.

3.2.4.4.4 Velocity Change Attitude Maneuver (Cont.)

When the "compare time to next event" coincides with the velocity change command, the engine valves are turned on, the spacecraft acceleration is integrated by counting down the pulses from programmer register until the velocity change desired is obtained. The "command done" signal from the programmer turns off the engine thru the switching assembly. Attitude is controlled thru the thrust vector control system while the engine is firing and is illustrated by Figure 3.2.4-10.

Upon completion of the velocity change, the programmer commands the reverse attitude maneuver sequence thru the same angles as initially commanded. Upon completion of the last maneuver, the spacecraft reacquires the Sun and Canopus and the gyros are switched back to the rate mode. The limit cycle deadbands are controlled by real time commands from the ground thru the programmer. The spacecraft then continues with "compare time to next event" in the programmer.

The velocity change maneuvers differ only in magnitudes of attitude and velocity change for midcourse corrections, lunar orbit injection, and orbit transfer at the moon.

Upon completion of all velocity change maneuvers, the N_2 pressurant shut-off squib is fired by the programmer thru the switching assembly.

3.2.4.4.5 Lunar Orbit Coast

Attitude control system operation while in lunar orbit proceeds in a similar manner as during the translunar phase with one important exception. Occultation of the Sun and Canopus by the Moon can be expected to occur each orbit. Absence of either the "Sun present" or "Canopus present" signal switches the appropriate gyro to inertial

POWERED FLIGHT ATTITUDE CONTROL

(Pitch and Yaw)

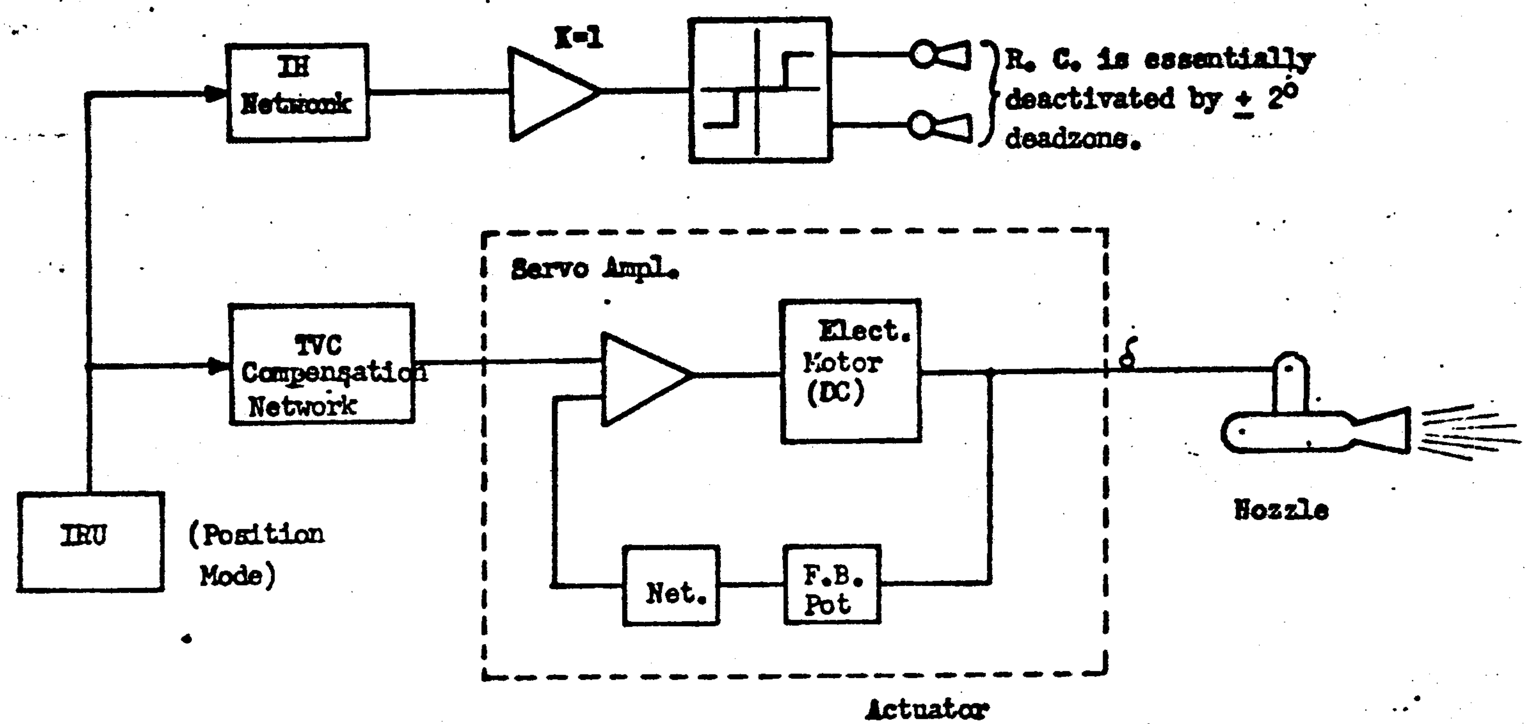


FIGURE 3.2.4-10

3.2.4.4.5 Lunar Orbit Coast (Cont.)

hold mode until the celestial reference is again present later in the orbit. The "present" signals automatically switch the sun and canopus sensor signals back into the reaction control system and the gyros to the rate mode.

3.2.4.4.6 Photography Maneuver

The proper sequence and magnitude of maneuvers to obtain photos of the desired targets are stored in the programmer for execution at the required time. The maneuver sequence proceeds in the same manner as for a velocity change attitude maneuver with an added step which permits the yaw channel of the attitude control system to accept a crab angle signal from the camera. Upon completion of the yaw maneuver commanded by the programmer, yaw control is switched to the V/H sensor crab angle signal and allowed 51.2 seconds to acquire. Then the yaw gyro is switched to inertial hold mode while the camera takes the photos. The reverse attitude maneuver sequence of yaw, pitch and roll is accomplished with subsequent reacquisition of the sun and Canopus. The system continues to operate in the limit cycle mode with the deadband selected by ground command for the remainder of the photo and readout phases of the mission. Figure 3.2.4-11 illustrates the photo mode yaw control.

PHOTO MODE YAW CONTROL

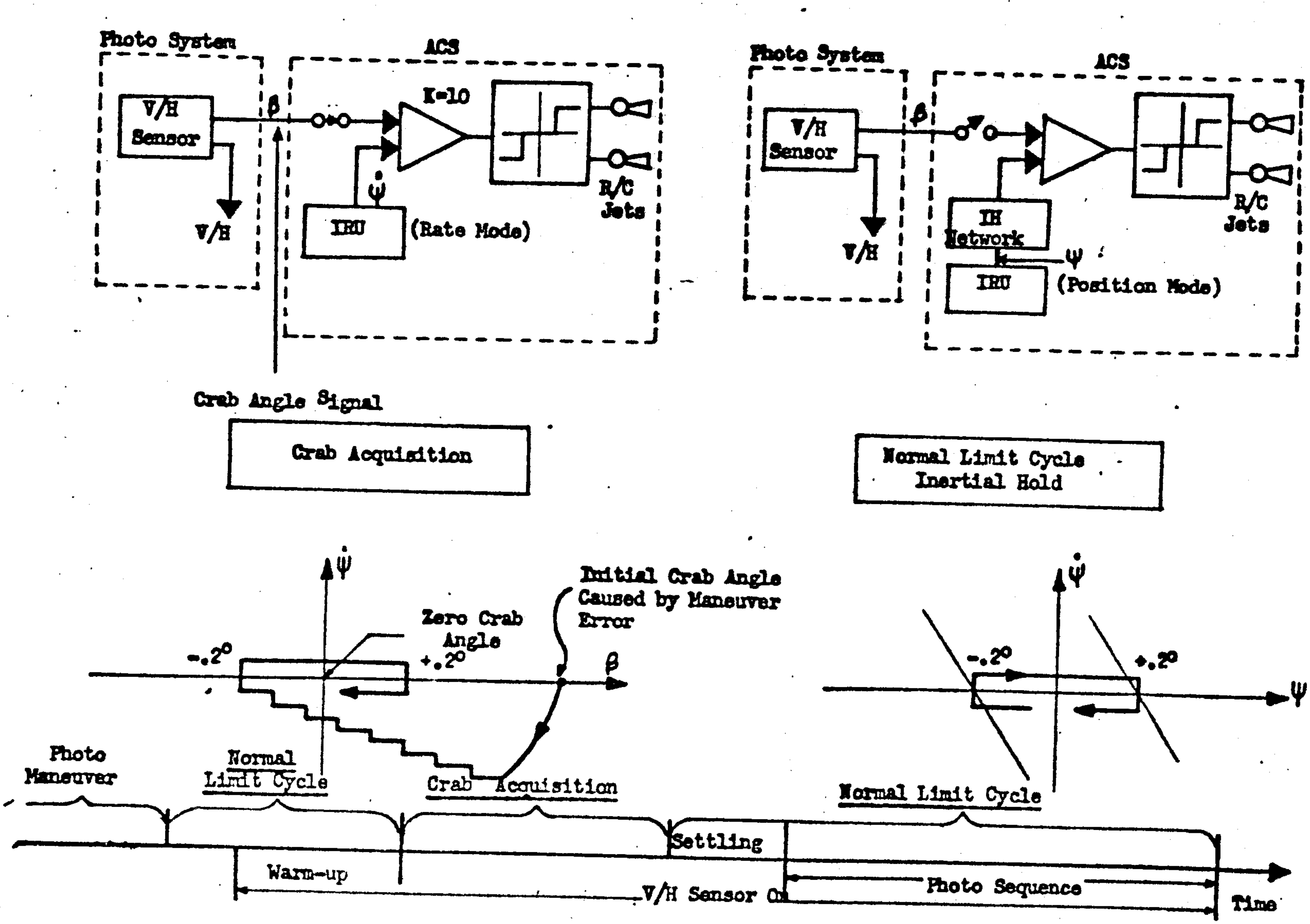


FIGURE 3.2.4-11

3.2.4.4.7 Extended Life

Upon completion of the photo mission period (30 days) the extended life configuration of the attitude control system is selected by ground command. Since the star tracker field of view is only 16° in yaw, and the excursion of the line of sight to Canopus may exceed this after 30 days (depending on time of year) the star tracker is turned off and the roll gyro switched to the inertial hold mode. The spacecraft continues to limit cycle ± 2° in roll for the remainder of the mission without a celestial reference. Also, to conserve nitrogen gas, the pitch and yaw fine sun sensors are switched off and the coarse sensors only switched on. This serves to widen the pitch and yaw deadbands to ± 12° while in the sun. When the sun is "not present" the inertial hold mode is automatically selected which also confines the spacecraft to ± 2° limit cycle for that portion of the orbit. The options available are illustrated by Figure 3.2.4-8. As an alternate mode of operation, roll control may be switched to the constant rate mode while the star tracker is off by supplying a false "canopus present" signal from the programmer. This alternate may be advantageous with respect to nitrogen consumption.

3.2.5 Velocity Control Subsystem

3.2.5.1 Function

After injection into the translunar trajectory by the Atlas/Agena launch vehicle, all spacecraft velocity change requirements will be furnished by the Velocity Control Subsystem. The nominal mission profile requires that up to four velocity change maneuvers be accomplished prior to attaining the final lunar orbit. In addition, it is desirable that additional maneuvers be possible, if needed, to modify the final orbit. The nominal mission maneuvers are:

3.2.5.1 Function (Cont.)

Midcourse maneuver for trajectory refinement; one or two maneuvers are planned.

Lunar orbit injection maneuver to place the spacecraft in a lunar orbit whose apolune is approximately 1850 km and whose perilune is approximately 250 km.

An orbit transfer maneuver to place the spacecraft into its final orbit for photographic reconnaissance. The orbit shall have an apolune of approximately 1850 km and a perilune of approximately 46 km.

3.2.5.2 Design Requirements

Accomplishment of the basic spacecraft mission imposes certain design requirements on the Velocity Control Subsystem. These requirements are summarized as follows:

- Operate and be storable in a space environment for up to 18 days.
- Provide a nominal total impulse capability of 73,000 lb-sec.
- Possess a minimum capability of:
  - Four engine operating cycles.
  - 710 seconds of operation.
  - Up to 614 seconds of operation in a single firing cycle.

As with any other spacecraft system, the Velocity Control Subsystem shall have high reliability and minimum weight. Target goals for the VCS are a reliability of 0.9926 and a weight (including propellants) of 324 lbs.

3.2.5.3 Subsystem Description

The design requirement for high reliability and the minimum program time from concept to flight dictate the employment of existing concepts and hardware wherever possible. This, together with the require-

3.2.5.3 Subsystem Description (cont.)

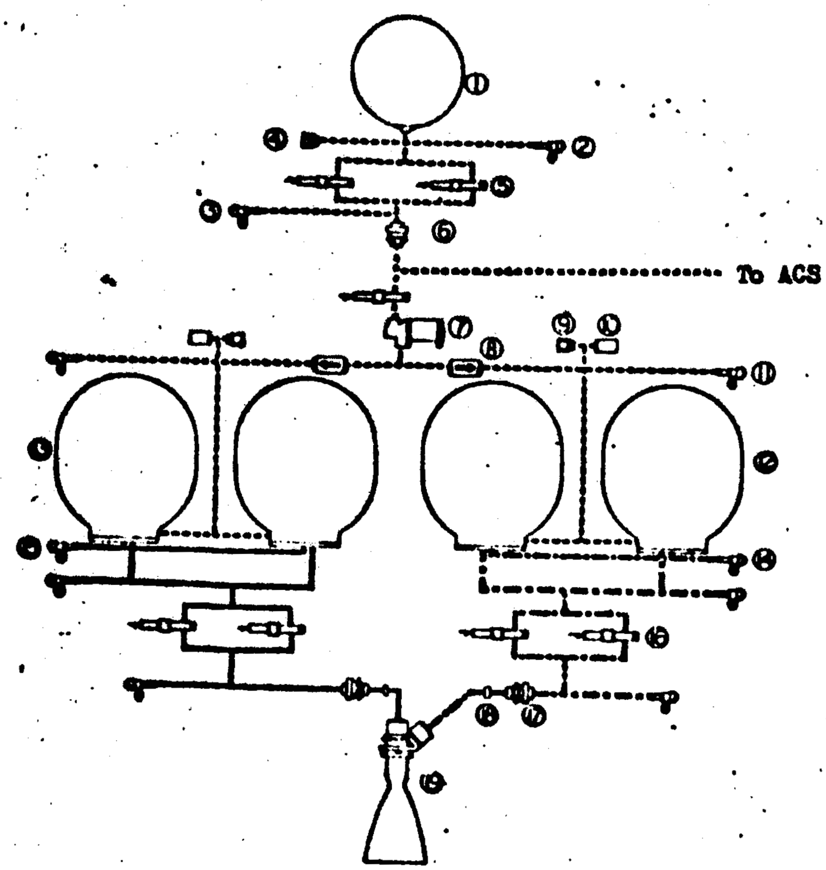
ments for relatively high performance and space storability, led to the selection of a storable, bipropellant hypergolic propellant combination.

The Velocity Control Subsystem employs nitrogen tetroxide as the oxidizer and Aerezine-50 (a 50-50 blend, by weight, of hydrazine and unsymmetrical dimethylhydrazine) as the fuel. The fuel and oxidizer are contained in two propellant tanks each to minimize movement of the spacecraft center of gravity. The propellants are positively expelled from their respective tanks by unheated gaseous nitrogen acting upon teflon bladders within the tanks. The nitrogen is stored in a titanium vessel at an initial pressure of 3500 psia; a pressure regulator reduces the supply pressure to the desired propellant tank pressure. The rocket engine is a 100 lbs. thrust, pressure-fed, radiation cooled engine that has been developed for the Apollo program (Service Module and LEM attitude control). The engine is mounted in 2-axis gimbals, and electromechanical actuators provide thrust vector directional control during engine operation (pitch and yaw control; roll control is maintained by the Attitude Control Subsystem nitrogen thrusters). Figure 3.2.5-1 shows the schematic arrangement of the various VCS components. As shown, the system also includes several valves, filters, orifices, and test points that are required for operation and/or test and checkout. Squib-actuated isolation valves maintain pressurization and propellant system integrity prior to initial operation.

3.2.5.4 Subsystem Operation

After spacecraft separation from the Agena, dual squib-actuated nitrogen isolation valves are opened. This operation admits high pressure

VELOCITY CONTROL SUBSYSTEM SCHEMATIC



	COMPONENT	QTY	SOURCE CONTROL DWG.	SUPPLIER	PROGRAM
1	STORAGE BOTTLE	(1)	25-50994	Boeing	
2	N_2 FILL VALVE	(1)	10-72026-1, -2	Firewel	
3	GROUND CHECKOUT	(1)	10-72026-1, -2	Firewel	
4	PRESSURE SENSOR	(1)	10-72004-3	Fairchild	Gemini
5	N_2 SQUIB VALVE	(2)	10-72017-1	OEA	
6	N_2 FILTER	(1)	10-72018-1	Vacco Valve	Appollo LEM
7	PRESSURE REGULATOR	(1)	10-70057-1	National Waterlift	Gemini
8	QUAD CHECK VALVE	(2)	10-72016-1, -2	APCO	Apollo SM
9	PRESSURE SENSOR	(2)	10-72004-2	Fairchild	Gemini
10	RELIEF VALVE	(2)	10-72023-1	Calmec	Apollo SM
11	TEST AND VENT VALVE	(2)	10-72026-1, -2	Firewel	
12	50/50 TANK	(2)	10-70055-1	Bell Aerosystems	Apollo CM
13	N_2O_4 TANK	(2)	10-70056-1	Bell Aerosystems	Apollo CM
14	50/50 FILL VALVE	(1)	10-72021-1, -2	J. C. Carter	Apollo CM
15	N_2O_4 FILL VALVE	(1)	10-72021-4, -8	J. C. Carter	Apollo CM
16	PROPELLANT SQUIB VALVE	(4)	10-72019-1	OEA	
17	PROPELLANT FILTER	(2)	10-72024-1	Vacco Valve	
18	TRIM ORIFICE	(2)			
19	ENGINE	(1)	10-70054-1	Marquardt	Apollo SM
20	GROUND CHECKOUT	(1)	10-72026-1, -2	Firewel	
21	TEMPERATURE SENSOR	(1)	10-72005-2	Gulton	Nimbus
22	SQUIB SHUTOFF VALVE	(2)	10-72071-1	OEA	

FIGURE 3.2.5-1

3.2.5.4 Subsystem Operation (Cont.)

nitrogen into the system, thereby pressurizing the propellant tanks and providing a source for the attitude control thrusters. A short time prior to the first midcourse maneuver the squib-operated propellant isolation valves are opened and propellant flows down to the engine solenoid valves. It is necessary to momentarily actuate the engine valves prior to the squib valves to bleed trapped gases from the propellant lines in order to insure proper propellant sequencing into the engine for the initial start. The system is now armed.

In response to on-board commands, the engine solenoid valves are opened and the VCS operates until the programmed velocity increment has been achieved. At that point, engine operation is terminated by removal of the engine solenoid control signal. The subsystem remains in an armed condition. Following the last propulsive maneuver, the normally-open, squib-operated nitrogen isolation valves are closed. This operation seals off the VCS for the remainder of the mission to minimize the loss of nitrogen which would otherwise be used for spacecraft attitude control.

3.2.5.5 Subsystem Performance

Table 3.2.5-1 summarizes pertinent Velocity Control Subsystem performance parameters.

TABLE 3.2.5-1
VCS PERFORMANCE

Thrust, lbs	100 ± 5
Specific Impulse, sec	
Nominal	276
Minimum	270
Mixture Radio, Oxidizer/Fuel	2.00 ± 0.076
Useable Propellant, lbs.	
Minimum	262
Maximum	267

3.2.5.6 <u>Constraints</u>

The only constraint imposed on the spacecraft by the VCS is that the temperature be maintained in a range of 35-85°F. Lower temperatures will result in propellant freezing, and higher temperatures will cause an increase in propellant tank pressure which, in turn, will cause relief valve operation and a loss of nitrogen.

3.2.6 <u>Structure and Mechanisms - Description</u>

The spacecraft structure consists of three decks and their supporting structure. The location of these decks and their associated station designation are as shown in Figure 3.2.6-1. For convenience in structural analysis the spacecraft structure was separated into two assemblies, the equipment mounting deck (Station 237) and the superstructure (Station 180 to Station 237).

The equipment mounting deck consists of a structural ring around the perimeter of a stiffened plate. As a structural assembly the equipment mounting deck provides:

- shear continuity between the spacecraft and the adapter,
- a load path for the compressive loads due to the V-band tension,
- support for the antennas and solar panels and for their deployment mechanisms, and
- support for equipment.

The primary truss structure (Station 237 to Station 202) is composed of six straight round tubes and an arch. These members are attached to the equipment mounting deck (Station 237) the tank deck (Station 202) such that pairs of tubes form three sides and the arch forms the fourth side of the primary truss. The deck truss attachments are spaced at 90 degrees on both the equipment and tank decks. At the equipment deck (Station 237) the attachments are rotated 45 degrees

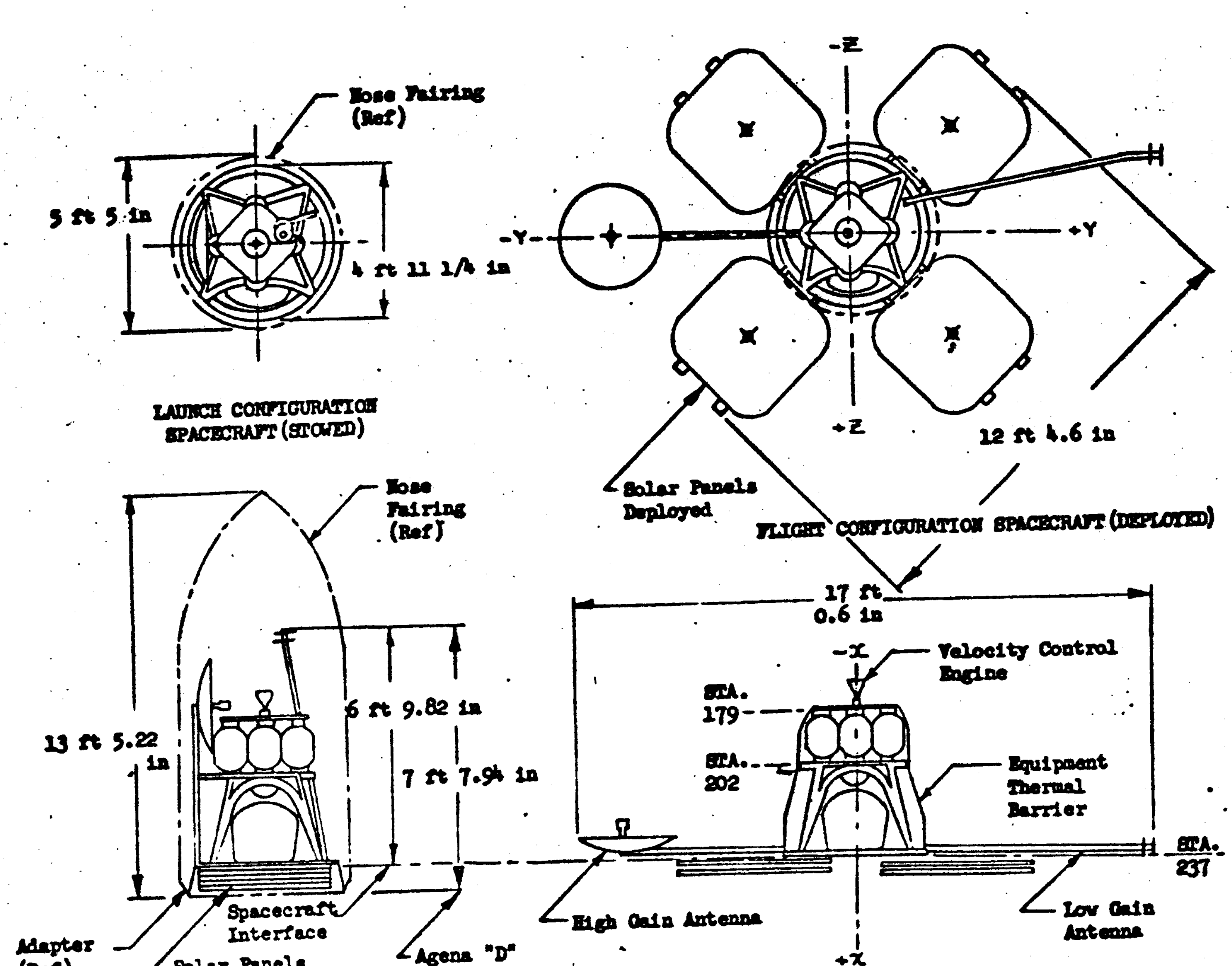


FIGURE 3.2.6-1

SPACECRAFT SIZE AND GENERAL ARRANGEMENT

3.2.6 Structure and Mechanisms - Description (Cont.)

from the spacecraft y and z reference axes. At the tank deck (Station 202) the attachments are in line with the necessary y and z reference axes.

The primary truss structure (Station 237 to Station 202) provides structural attachment between the equipment mounting and tank decks. The tank deck (Station 202) is a machined ring, v-shaped in cross section, closed out with a flat sheet. The tank deck has attachments for the primary truss, the support structure truncated cone, the fuel and oxidizer tanks and the nitrogen tank. It also provides for the support of the thermal barrier, the high gain antenna, the meteoroid detector array, and the switching assembly (black box).

The support structure (Station 202-180) consists of a truncated cone and cylinder with cutouts as required to provide for installation and operation of Velocity Control System components. It provides structural attachment between the engine deck and tank deck, support of tubing and wiring, and support of Velocity Control System Components.

The engine deck (Station 179) is a bead stiffened plate supported by a ring at the upper end of the support structure cylinder. The engine deck provides mounting provisions for the velocity control engine, and component mounting provisions for the attitude control thrusters and for the remote sun sensors, support structure and release mechanism for the omni antenna, and provision for attachment of ground handling equipment.

The deployment mechanisms and ordnance device locations are illustrated in the Figures 3.2.6-2 through 3.2.6-5.

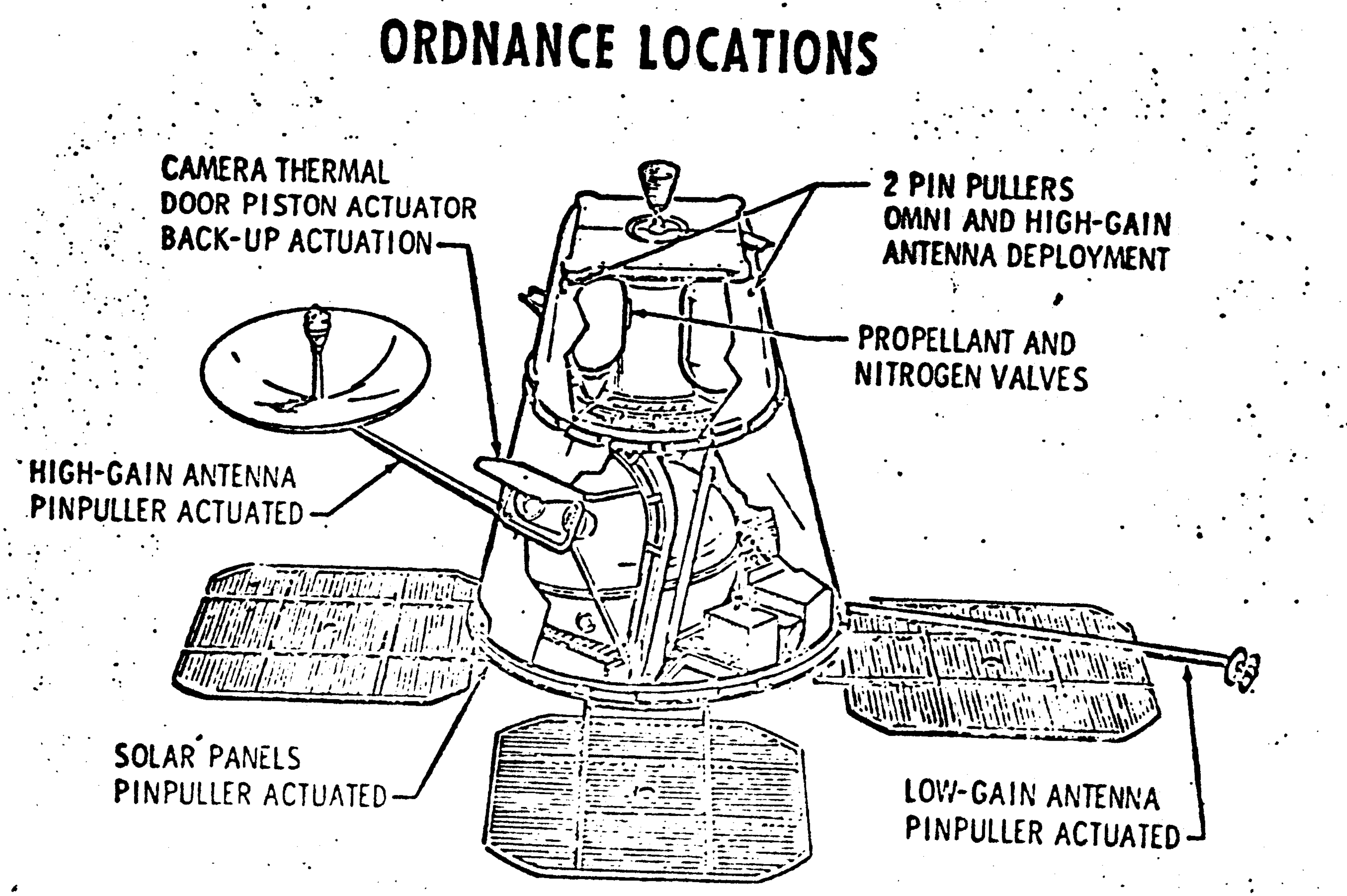

ORDNANCE LOCATIONS
CAMERA THERMAL
DOOR PISTON ACTUATOR
BACK-UP ACTUATION
2 PIN PULLERS
OMNI AND HIGH-GAIN
ANTENNA DEPLOYMENT
PROPELLANT AND
NITROGEN VALVES
HIGH-GAIN ANTENNA
PINPULLER ACTUATED
SOLAR PANELS
PINPULLER ACTUATED
LOW-GAIN ANTENNA
PINPULLER ACTUATED

FIGURE 3.2.6-2

SOLAR PANEL

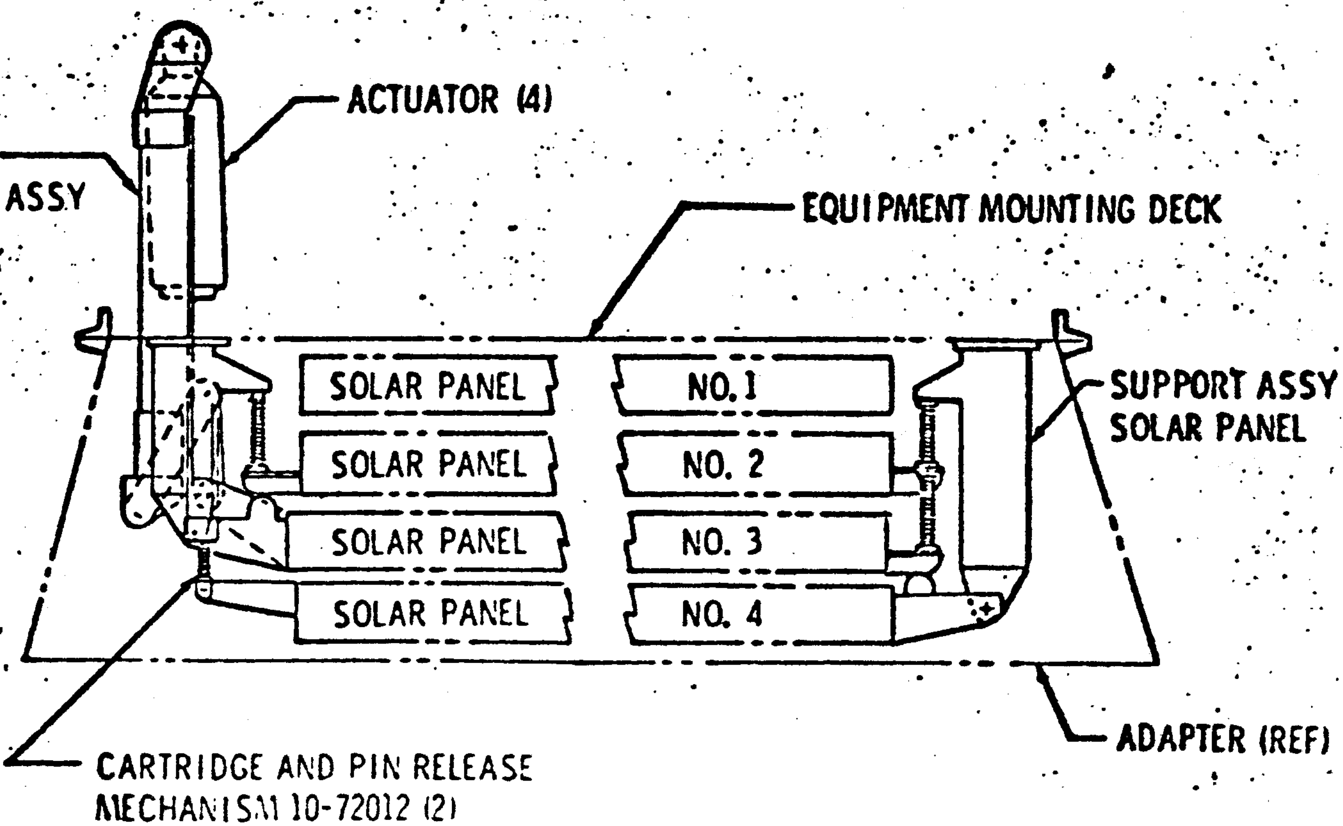


FIGURE 3.2.6-3

HIGH GAIN ANTENNA

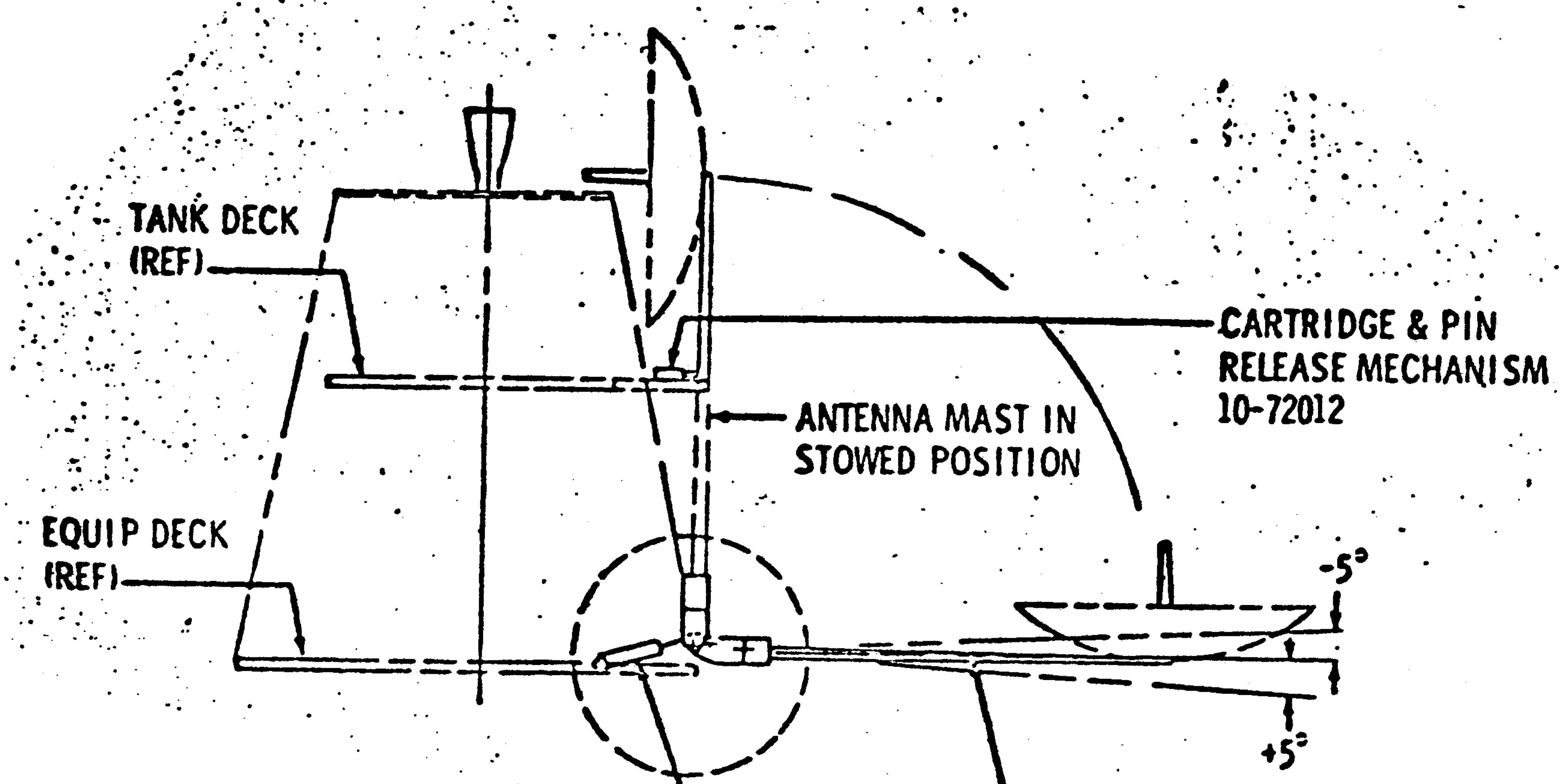


FIGURE 3.2.6-4

LOW GAIN ANTENNA

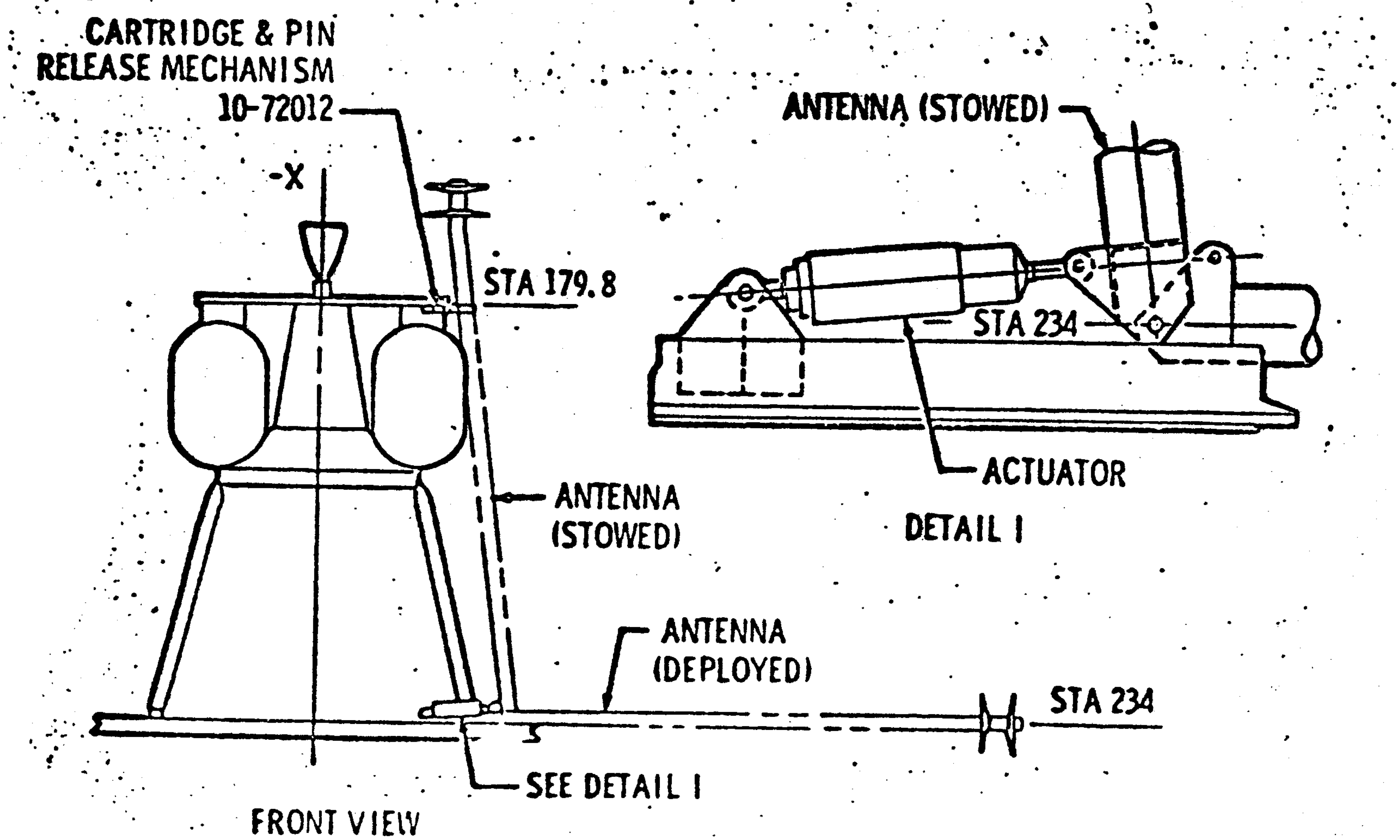


FIGURE 3.2.6-5

3.2.7 Thermal Control Subsystem

3.2.7.1 Concept

The fundamental concept for thermal control of the spacecraft is a completely passive system. This is accomplished by providing insulated boundaries on all sides of the spacecraft except one. The one non-insulated side of the spacecraft Equipment Mount Deck (EMD) is oriented toward the sun and temperature controlled by a special paint coating. The spacecraft components are cooled by using the EMD as a heat sink. All external components are thermally isolated from the internal spacecraft components. The external components are temperature controlled by special coatings.

3.2.7.2 Subsystem Requirements

The Lunar Orbiter prelaunch requirements are to maintain the EMD at 70° F. or below at any time there is power being dissipated in the spacecraft. Cleanliness and humidity control must also be maintained.

The spacecraft is thermally required to maintain passive temperature control for the following flight conditions:

Launch and earth park

Translunar

Initial Orbit - standby power (106 W)

Final Orbit - standby power (106 W)

Final Orbit - photo process power (159 W)

Final Orbit - photo readout power (223 W)

The thermal control requirements for final orbit are to control spacecraft temperature when the spacecraft is subjected to the tolerance boundaries shown in Table 3.2.7-1.

TABLE 3.2.7-1
ORBITAL TOLERANCE BOUNDARIES

ORBITAL CONDITION	APOLUNE KM	PERILUNE KM	SOLAR CONST. W/FT^2	EMD SOLAR ABSORB-TIVITY	ILLUMIN-ATION ANGLE DEG.	TARGET LAT. DEG.	LIGHT TIME HRS.	DARK TIME HRS.	ORBIT PERIOD HRS.	S/C POWER WATTS
Hot Extreme	2100	46	135	.25	90	0	2.99	.73	3.72	223
Nominal	1850	46	130	.22	60	-10	2.56	.92	3.47	106
Cold Extreme	1600	46	125	.21	37	-10	2.26	.97	3.23	106

The interior spacecraft components must be thermally coupled so as to maintain reasonable temperatures when power loads are cycled and when the spacecraft is cycled in and out of the shadow of the moon.

All external components must be provided with coatings that will control temperatures within the temperature limits of the materials.

3.2.7.3 System Description

The EMD temperature is controlled by coating the exterior surface with a silicone base paint using zinc-oxide pigment. The initial properties are:

- Emmissivity = .90
- Initial Absorbtivity = .22 ± .01

The maximum solar absorbtivity increase due to ultra violet is .02. The spacecraft temperature range during the translunar phase is 40^oF to 100^oF. During the final orbit phase they are as shown in Table 3.2.7-2.

TABLE 3.2.7-2
TEMPERATURE RANGES

FINAL ORBIT	HOT EXTREME °F.	COLD EXTREME °F.
Area Around TWTA	95	-12
Area Around Batteries	90	20
Area Around IRU	100	25
Area Under Camera	85	-6

NOTE: Changes in power density (W/FT^2) on the EMD would change the above temperatures.

The internal parts of the spacecraft are thermally coupled by radiation by painting all internal parts with a highly emmissive paint. The two exceptions are the Photo Subsystem and the Thermal Barrier (Figure 3.2.7-1) which are highly reflective. The Photo Subsystem is highly reflective to permit adequate temperature control of critical components inside of the Photo Subsystem. The Thermal Barrier is highly reflective because it was convenient and heat flow through it is extremely small so the emmissivity and reflectivity has very little effect on the spacecraft thermal control.

The external surfaces have thermal properties shown in Table 3.2.7-3.

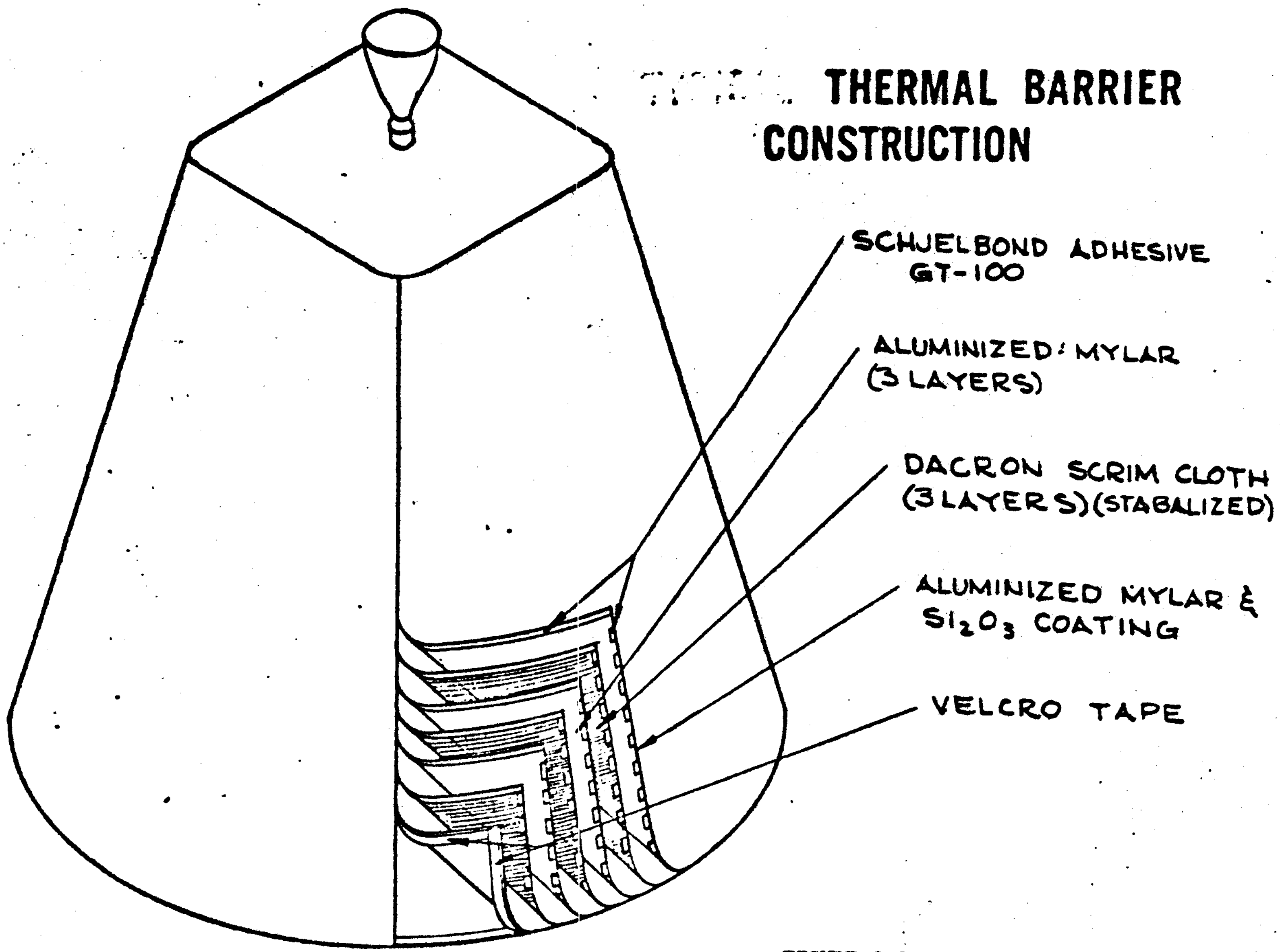

THERMAL BARRIER
CONSTRUCTION
SCHJELBOND ADHESIVE
GT-100
ALUMINIZED MYLAR
(3 LAYERS)
DACRON SCRIM CLOTH
(3 LAYERS)(STABALIZED)
ALUMINIZED MYLAR &
Si_2O_3 COATING
VELCRO TAPE

FIGURE 3.2.7-1

TABLE 3.2.7-3
SURFACE FINISH THERMAL PROPERTIES

	SOLAR ABSORPTIVITY / INFRA-RED EMMISSIVITY	EQUILIBRIUM TEMP. °F.
Thermal Barrier	1.0 or less	250 or less
Heat Shield	2.6 to 3.9	425 to 530
Low Gain Antenna	.22 to .28	30 to 60
High Gain Antenna	.20 to .28 Gear Box - 1.0 to 1.2 Dish = 1.0 to 1.2	30 to 60 250 to 280 250 to 280
Solar Arrays	.85 to 1.0	225 to 250
Rocket Nozzle	1.0 to 1.25	250 to 285

Equilibrium temperature is the steady state temperature of the infinitely thin item facing directly toward the sun and perfectly insulated on the back side. The actual temperature of each component is a function of heat flow to and from the surface facing the sun and the surfaces facing away from the sun. All external components are designed to be capable of withstanding full sunlight for an unlimited time period.